JET PROPULSION

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AMERICAN ROCKET SOCIETY

Research and Development

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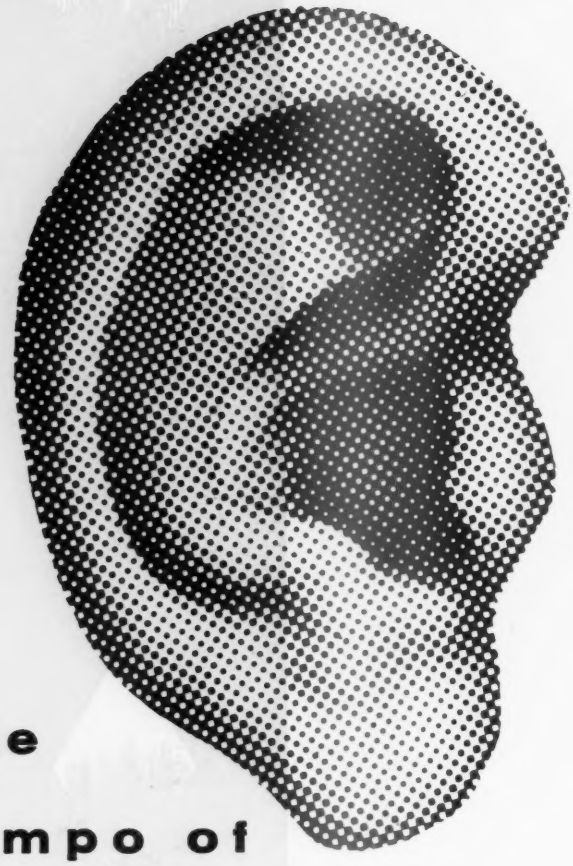

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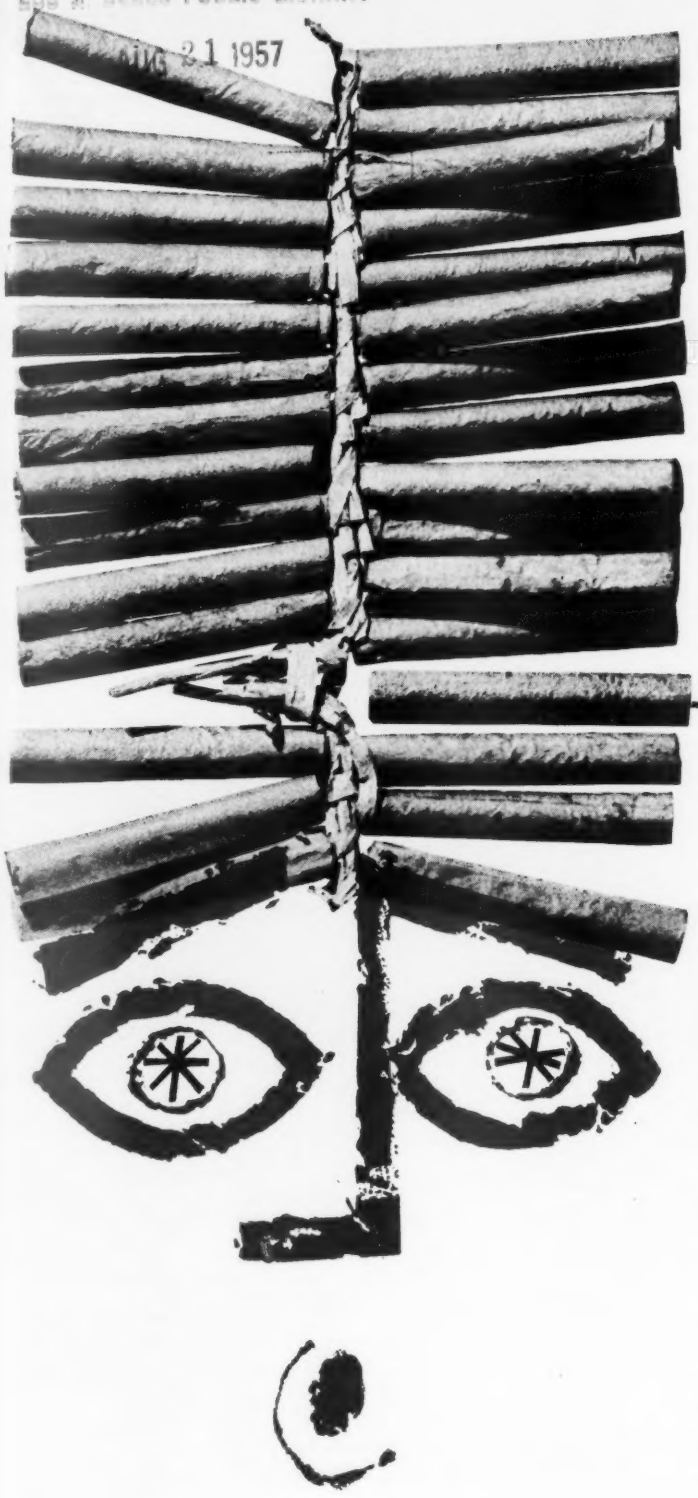

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Scope of JET PROPULSION

This Journal is a publication of the American Rocket Society devoted to the advancement of the field of jet propulsion through the dissemination of original papers disclosing new knowledge or new developments. As used herein, the term "jet propulsion" embrances all engines that develop thrust by rearward discharge of a jet through a nozzle or duct, and thus it includes air-consuming engines and underwater systems as well as rockets. JET PROPULSION is open to contributions dealing not only with propulsion but with other aspects of jet-propelled flight, such as flight mechanics, guidance, telemetering, and research instrumentation. Increasing emphasis will be given to the scientific problems of extraterrestrial flight.

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Even more complex than ground-to-air marksmanship is air-to-air gunnery, where opposing planes top 1000 mph, and fire missiles that move twice that fast. It's an entirely new science, based on principles first expounded by a Douglas scientist, and proved in repeated successful tests—even against unseen targets.

Dramatic moment in aviation history, a Douglas-built Nike-Ajax bores in on a drone bomber.

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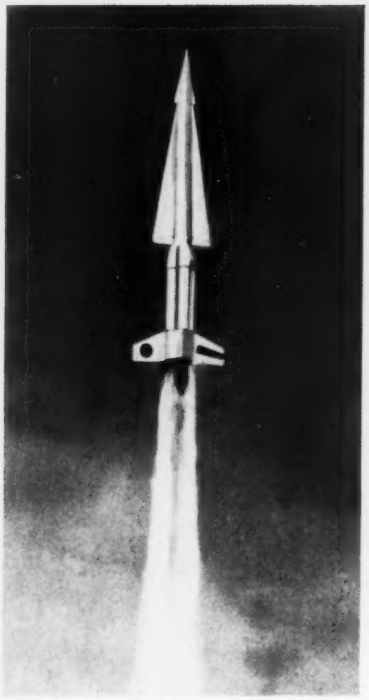

Here is Nike-Hercules, soon to take over defense of U. S. cities from the earlier Nike-Ajax. Tripling the range of its predecessor, Nike-Hercules can be armed with a nuclear warhead, to knock out entire fleets with a single blast.

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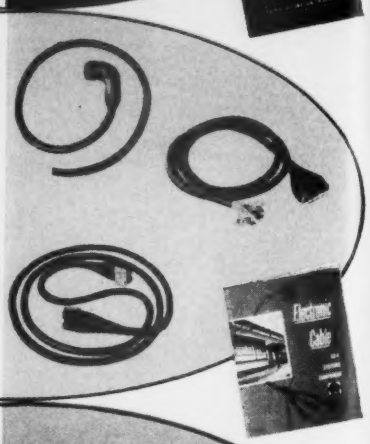
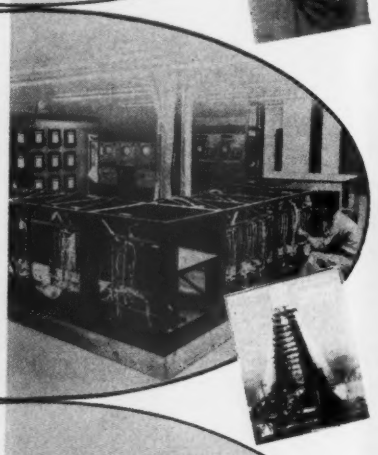
CABLE AS A COMPONENT

"Electronic Cable as a Systems Component" Bulletin No. 656. Generously illustrated pamphlet describes design and production of electronic cable in bulk and assemblies. Connectors and hardware; cable breakouts; environmental requirements; voltage rating, testing; cable procurement data—are some of the subjects covered in this bulletin.



CABLE SYSTEMS

"Engineered Cable Systems" Bulletin 357. Illustrated brochure describing Pacific Automation Products' pioneering in the use of the Systems Concept to coordinate cable design with the development of complex electronic systems. Liaison engineering teams cooperating with systems design groups establish cable as integral part of system, supervising cable construction concurrently with development of other elements.



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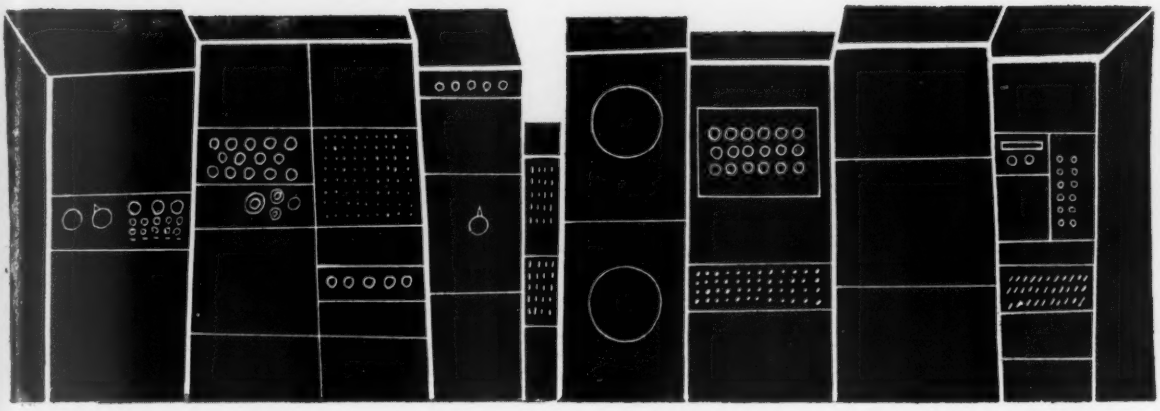

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Some Aspects Pointing Up Engineering Opportunities in Small Aircraft Gas Turbine Engineering

FROM STOKE'S LAW for the very small to Hoyle's work with the very large, the "3/2 power" principle maintains its validity. For an example in the middle range of our immediate experience, note the curves on the chart (1).

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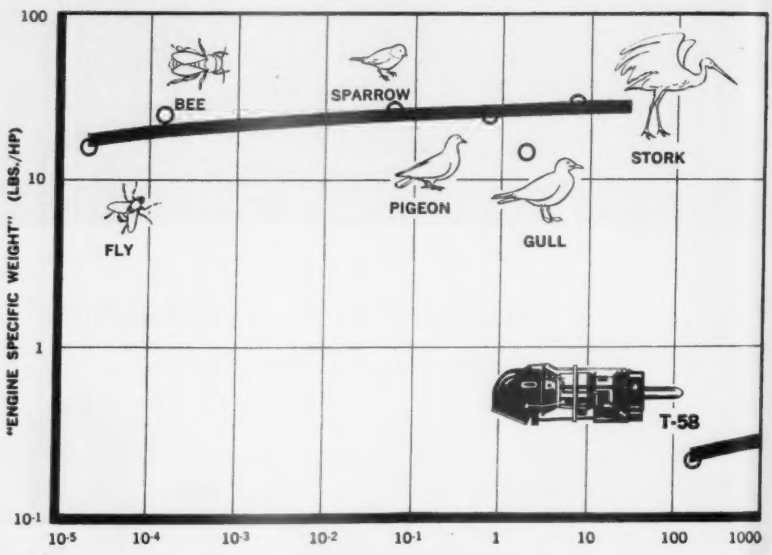
For the engineer who is specially curious about the small curve on the plot, we can recommend no better course for *him* than to contact Roger Hawk of General Electric's Small Aircraft Engine Department.

Engineers at SAED are busy extending the 3/2 principle to AGT applications. The success with which they have met in developing *small* highly efficient power plants (2) has created many new openings for engineers with strong backgrounds in AGT work.



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If you feel that you would like to put your degree and AGT experience (3) to work in the exacting region of that small lower curve, you can get additional details on these positions from Mr. Hawk.

NOTES AND REFERENCES

1. Values for the "engine specific weight" of the insects and birds were derived from Thompson . . . cf. "Growth and Form," 2nd Ed., 1942, Cambridge University Press; Chapter II on magnitude, especially pp. 45, 47. Birds were assumed to fly at 30 mph, insects at 5 mph.

2. The T58 Turboshaft — 250 lbs. without reduction gear, 1050 HP — now in production. The T64 Convertible — for turboprop or turboshaft use, depending upon gearing — now in development; the J85 Turbojet — details still classified.

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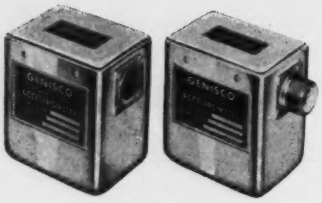

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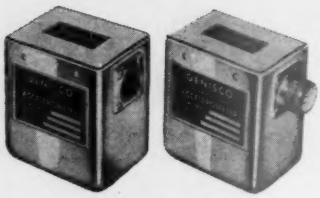

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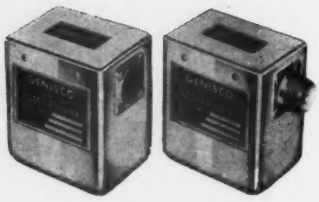

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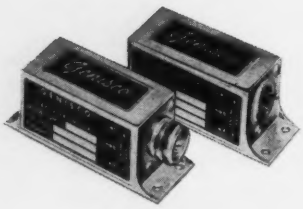

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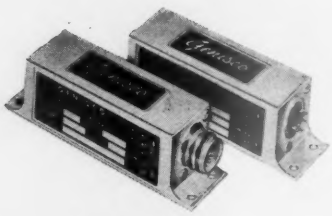
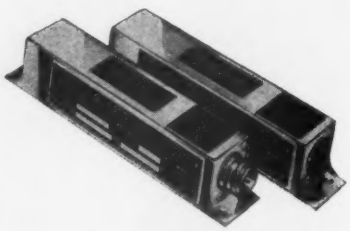


Model GMT

Basically a Model GMO with internal thermostat-operated heater, assuring maximum environmental stability within the instrument. Damping remains constant with change in ambient temperature.



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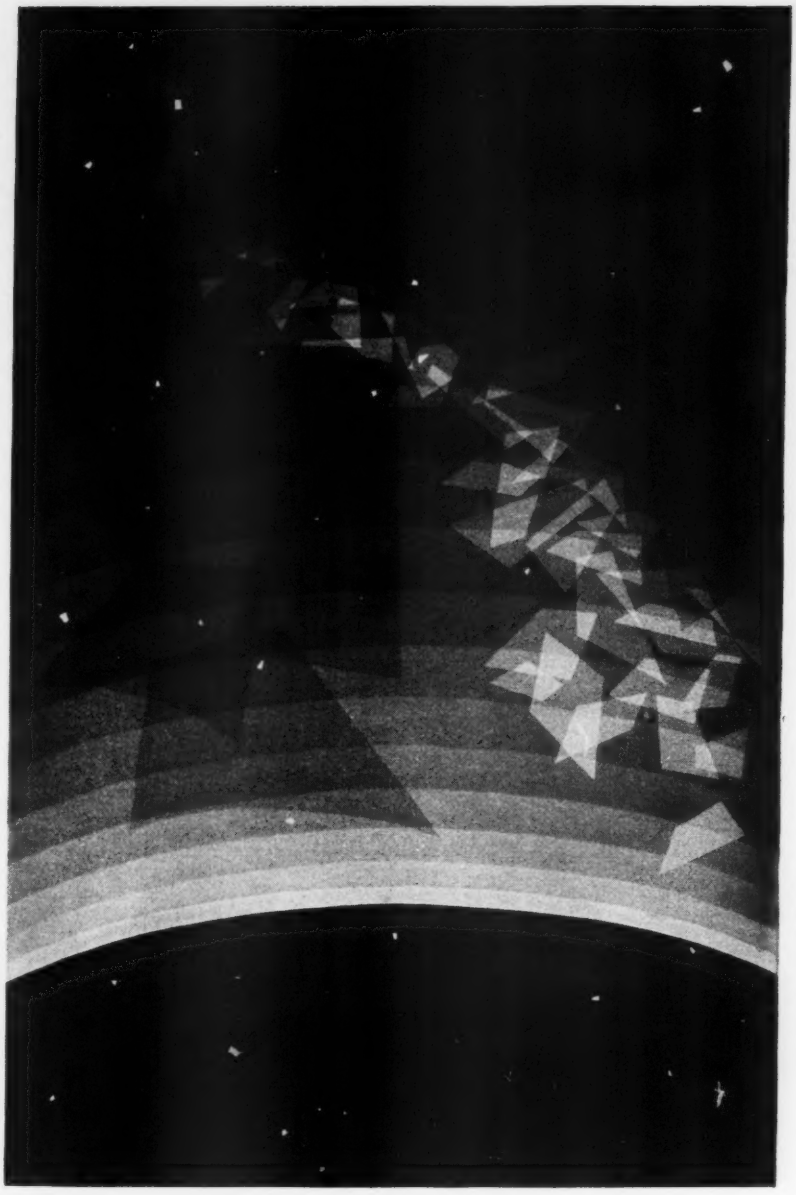

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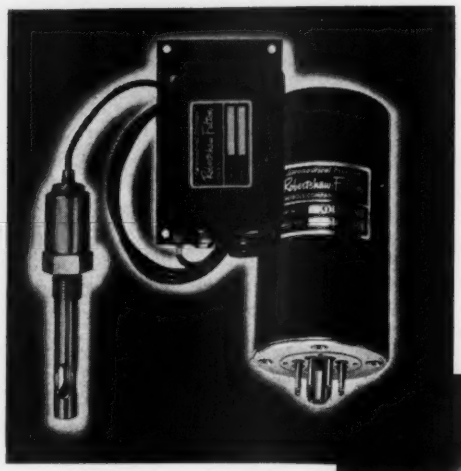

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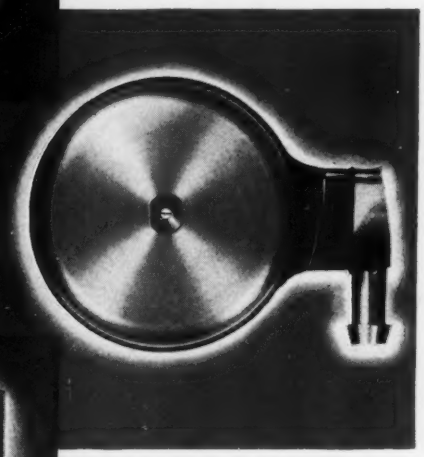

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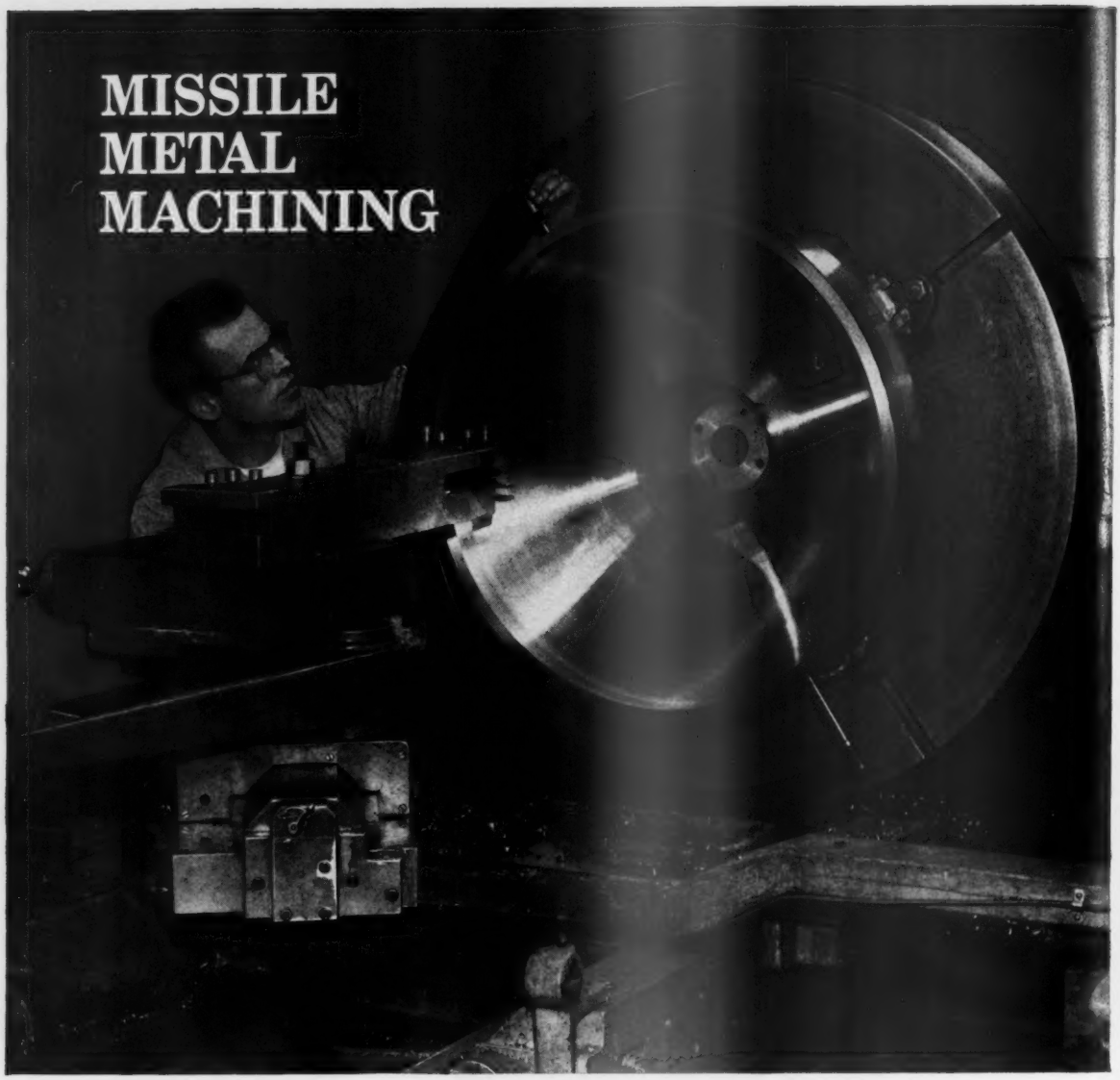

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Optimum Staging Technique for Multistaged Rocket Vehicles

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In the preliminary design of multistaged rocket vehicles the question always arises as to whether the staging technique used is optimum. The purpose of this paper is to present a preliminary design method of determining the optimum staging under a known set of conditions. The simplifying assumptions that have been made are not too restrictive in that they are in line with most preliminary analyses. The results show that it is possible with a very few calculations to predict the optimum performance of a multistaged rocket vehicle. This procedure would be applicable to the design of ballistic missiles and satellite vehicles.

Nomenclature

a_n = maximum acceleration for the nth stage
C = exhaust velocity = Ig
g = acceleration due to gravity (32.2 ft/sec^2)
I = specific impulse
m = mass
N = number of stages
t = time (seconds)
Δt_n = burning time for nth stage
T_n = thrust of nth stage
V = velocity
ΔV_n = velocity increment of nth stage
W_{on} = gross weight of nth stage
W_{pn} = propellant weight of nth stage
W_{en} = empty weight of nth stage
W_L = launch weight
W_{pay} = payload weight
ϵ_n = structural efficiency of nth stage
f_n = propellant loading factor of nth stage
λ_n = payload ratio of nth stage
τ_n = nondimensional burning time of nth stage
η = Lagrangian multiplier
μ_n = nondimensional velocity increment of nth stage
μ_T = nondimensional velocity of final vehicle
ω_n = fuel flow of nth stage

Introduction

IN THE preliminary design of multistaged rocket vehicles the problem arises as to what is the optimum staging technique. The usual approach to the problem is to determine the necessary launch weight to achieve a given end velocity for various combinations of staging times. Therefore, there is a need for a quick preliminary design technique to determine optimum burning times for each stage of a multistage vehicle.

The usual data that are available for a preliminary analysis are (a) initial flight speed of final vehicle, (b) specific impulse of rocket engines, (c) maximum acceleration for which structure will be designed and (d) payload weight. From these data the minimum launch weight, fuel load and structural weight are determined in order to give the required performance. In order to investigate the effect of payload weight, maximum acceleration, etc., a large number of calculations will have to be made. Therefore, the purpose of this paper is to present a method whereby the optimum staging technique can be determined by a few short calculations.

Assumptions

The basic assumptions that have been made in this analysis will be given below and discussed in some detail.

1. *Constant thrust during any given stage:* This assumption is not too restrictive since rocket engines are usually designed to operate at full power at all times. The variation of the thrust with altitude is usually small.

2. *Constant specific impulse:* The specific impulse will depend primarily on the engine design and the type of fuel used. Therefore, the assumption of constant specific impulse is very close to reality, except at low altitudes where it will decrease slightly.

3. *Zero drag:* It has been shown that in the case of large vehicles the effect of drag on the boost phase is very small (1).[2]

4. *Constant g:* The effect due to the variation in g should be small (e.g., the difference between the value at 300,000 ft and sea level is only 3 per cent).

Method of Analysis

It it is assumed that the flight path is vertical, the equation governing the motion of the vehicle is given by

$$-mg = m\frac{dV}{dt} + C\frac{dm}{dt} \quad \ldots\ldots\ldots\ldots [1]$$

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[1] Senior Aerodynamics Engineer.

[2] Numbers in parentheses indicate References at end of paper.

If a notation is used similar to that of (1), Equation [1] can be integrated and written as

$$\Delta V_n + g\Delta t_n = Ig \ln \frac{W_{on}}{W_{on} - W_{pn}} \quad \text{[2]}$$

We can choose a nondimensional velocity and time that will eliminate the specific impulse as a parameter and also simplify the equations.

$$\mu_n = \frac{\Delta V_n}{Ig} \qquad \tau_n = \frac{\Delta t_n}{I} \quad \text{[3]}$$

Equation [2] can be rewritten as

$$\mu_n + \tau_n = \ln \frac{1}{1 - \zeta_n} \quad \text{[4]}$$

where $\zeta_n = W_{pn}/W_{on}$.

Equation [4] gives a relationship for the nondimensional velocity increment and burning time in terms of the propellant loading factor for each stage.

The results of (1) can be summarized by

$$\frac{W_{on}}{W_{on+1}} = \frac{1 - \epsilon_n}{1 - \zeta_n - \epsilon_n} = \lambda_n \quad \text{[5]}$$

Making the following definitions

W_L = launch weight
W_{pay} = weight of final vehicle

then

$$\frac{W_L}{W_{o2}} \cdot \frac{W_{o2}}{W_{o3}} \cdots \frac{W_{oN}}{W_{pay}} = \lambda_1\lambda_2 \ldots \lambda_N = \frac{W_L}{W_{pay}} \quad \text{[6]}$$

Equation [6] gives the ratio of launch weight to payload weight in terms of the propellant loading factor and the structural efficiency of each stage. The problem now is to relate the payload ratio to the burning time.

If the thrust of each stage is defined as

$$T_n = I\omega_n = I \frac{W_{pn}}{\Delta t_n} = \frac{W_{pn}}{\tau_n} \quad \text{[7]}$$

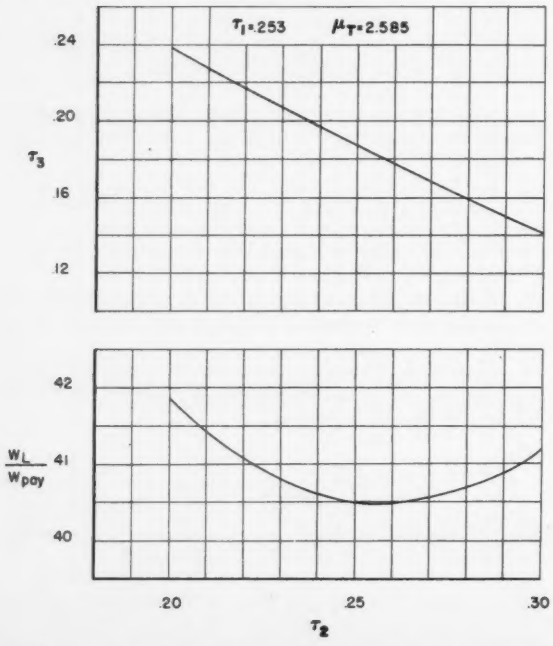

Fig. 1 Off-optimum design conditions for sample problem

Table 1 Sample calculation

V = 25,000 fps	I = 300 sec	μ_T = 2.585
a_1 = 8	a_2 = 10	a_3 = 8
ϵ_1 = 0.05	ϵ_2 = 0.05	ϵ_3 = 0.10

τ_1	τ_2	τ_3	$\mu_{T_{cal}}$
0.15	0.160	0.0950	1.911
0.20	0.203	0.1320	2.269
0.25	0.257	0.1760	2.572
0.253	0.259	0.1785	2.585

or

$$\frac{T_n}{W_{on}} = \frac{\zeta_n}{\tau_n} \quad \text{[8]}$$

the maximum acceleration for a constant thrust stage will occur at the end of the burning time and can be written as

$$a_n = \frac{T_n}{W_{on+1} + W_{en}} = \frac{W_{on}}{W_{on+1} + W_{en}} \frac{\zeta_n}{\tau_n} \quad \text{[9]}$$

or

$$a_n\tau_n = \frac{\zeta_n}{1 - \zeta_n} \quad \text{[10]}$$

Then it can be shown that

$$\ln (1 + a_n\tau_n) = \ln \frac{1}{1 - \zeta_n} \quad \text{[11]}$$

or from Equation [8]

$$\ln (1 + a_n\tau_n) = \mu_n + \tau_n \quad \text{[12]}$$

If the nondimensional final velocity is defined

$$\mu_T = \frac{V}{Ig} \quad \text{[13]}$$

then

$$\mu_T = \sum_{n=1}^{N} \mu_n = \sum_{n=1}^{N} \ln (1 + a_n\tau_n) - \sum_{n=1}^{N} \tau_n \quad \text{[14]}$$

Equations [6, 14] define the problem, and the variables that have been assumed known are a_n, ϵ_n, and μ_T; hence, the problem that remains is to find the τ_n's that will give the minimum W_L/W_{pay} and will satisfy Equation [14]. The minimal solution is one that will satisfy the following equations (2)

$$\frac{W_L}{W_{pay}} \frac{1}{\lambda_1} \frac{\partial\lambda_1}{\partial\tau_1} + \eta \frac{\partial\phi}{\partial\tau_1} = 0 \quad \text{[15-1]}$$

$$\frac{W_L}{W_{pay}} \frac{1}{\lambda_2} \frac{\partial\lambda_2}{\partial\tau_2} + \eta \frac{\partial\phi}{\partial\tau_2} = 0 \quad \text{[15-2]}$$

$$\frac{W_L}{W_{pay}} \frac{1}{\lambda_N} \frac{\partial\lambda_N}{\partial\tau_N} + \eta \frac{\partial\phi}{\partial\tau_N} = 0 \quad \text{[15N]}$$

$$\phi = 0 \quad \text{[15N + 1]}$$

where

$$\phi = \sum_{n=1}^{N} \ln (1 + a_n\tau_n) - \sum_{n=1}^{N} \tau_n - \mu_T$$

η = Lagrangian multiplier

The solution of Equations [15-1 to 15N] is

$$a_1\epsilon_1 \left[\tau_1 - \frac{1 - \epsilon_1}{a_1\epsilon_1}\right]\left[\tau_1 - \frac{a_1 - 1}{a_1}\right] = a_n\epsilon_n \left[\tau_n - \frac{1 - \epsilon_n}{a_n\epsilon_n}\right]\left[\tau_n - \frac{a_n - 1}{a_n}\right] \quad \text{[16]}$$

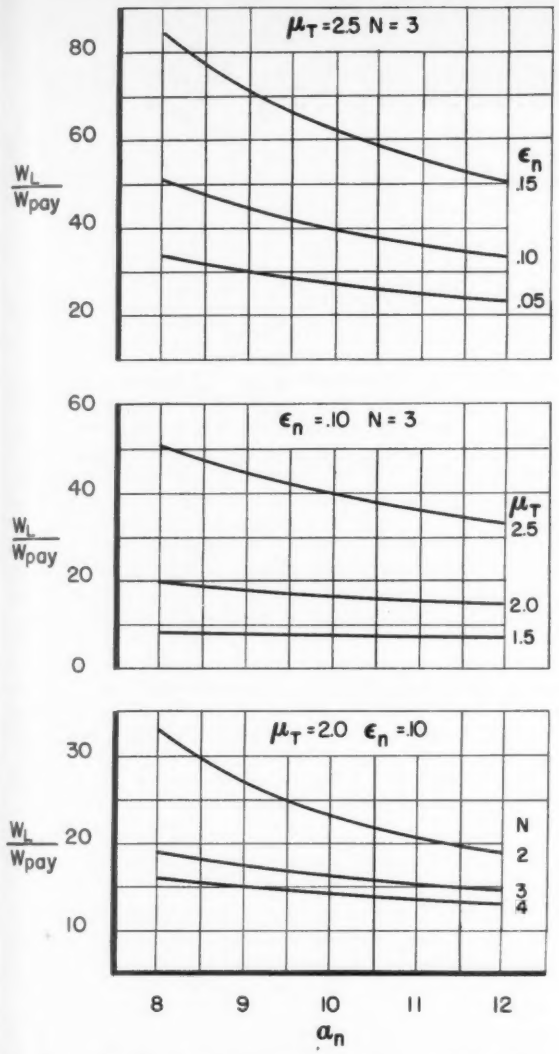

Fig. 2(a) **Parametric studies for similar stages—varying maximum acceleration**

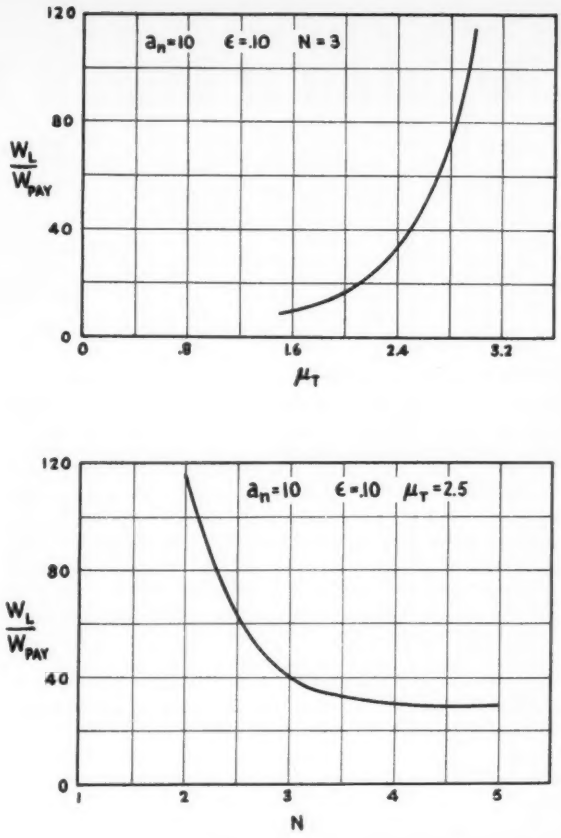

Fig. 2(b) **Parametric studies for similar stages—constant maximum acceleration and structural efficiency**

From Equation [16] the burning time of any stage can be determined in terms of the burning time of the first stage. Along with equation [15N + 1] we can determine the optimum staging technique.

Results

The solution of Equations [15N + 1 and 16] can only be done by trial and error. If a value for τ_1 is chosen, then τ_2, τ_3, ..., τ_N can be computed. Since Equation [16] is a quadratic in τ_N there will be two solutions for τ_n; the correct solution will be the smallest positive root. Once all of the burning times are computed, a value for μ_T can be calculated; if this calculated value does not agree with the actual value, the procedure is continued until the calculated and the actual value agree. A sample calculation is shown in Table 1. The results for this set of conditions are shown in Table 2.

In Fig. 1 a comparison is shown for the off-optimum design conditions. It can be seen that much can be gained by taking care in the choice of the staging technique. In Fig. 2 the effect of various parameters is shown for the case where the maximum acceleration and structural efficiency are the same for all stages.

References

1 Malina, F. J., and Summerfield, M., "The Problem of Escape from the Earth by Rocket," *Journal of the Aeronautical Sciences*, vol. 14, Aug. 1947, pp. 471–480.

2 Hildebrand, F. B., "Methods of Applied Mathematics," Prentice-Hall, New York, 1952, pp. 120–122.

Table 2 Final results for sample problem

Payload weight = 1000 lb

Stage	τ_n	λ_n	Δt_n	W_{pn}	W_{en}	W_{on}	T_n	V_{final}
1	0.253	3.39	76	27,153	1427	40,500	107,100	8,260
2	0.259	4.15	78	8,617	453	11,950	33,200	18,120
3	0.1785	2.88	54	1,692	188	2,880	9,500	25,000

The Stability of Bodies of Revolution at Very High Mach Numbers

A. F. CHARWAT[1]

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The Newtonian theory is used to study the stability of bodies of revolution at high hypersonic velocities such as may be encountered in the flight of satellites or the reentry of high altitude missiles. The important stability criteria are brought out and the allowable approximations are studied. Several numerical examples are worked out. It is observed that configurations with cylindrical afterbodies exhibit two possible equilibrium trajectories. One is a vertical dive and the second is a glide at an angle uniquely determined by the location of the center of gravity. The latter is the only completely stable trajectory. This is brought about by the characteristic of the Newtonian wake and it can be eliminated by using a truncated cone instead of a cylinder for the afterbody. The importance of the drag force in determining the stability of bluff bodies is shown.

Introduction

IN THE case of very large Mach numbers and values of the isentropic exponent approaching one, the surface pressure on an aerodynamic body can be predicted on the basis of the Newtonian impact theory (1).[2] Although the theory is approximate, its simplicity makes it an attractive tool for preliminary investigation.

The Newtonian concept assumes that the shock wave lies on the surface of the body. The normal component of momentum of the flow is lost upon impact, and the tangential component remains unchanged. In consequence, the local pressure coefficient depends only on the local slope of the surface with reference to the relative wind. A refinement which accounts for the centrifugal forces in the disturbed flow is available (2, 3), but the magnitude of the correction is not very large (8).

A recent study (5) of the accuracy of the Newtonian theory (as well as other approximate analyses) in predicting the pressure on nonlifting bodies of revolution indicates that for values of the hypersonic similarity parameter greater than 2 the error in the pressure drag, as calculated by that method, is acceptable. Calculations were not carried out for values of the parameter greater than 2, but an extrapolation of the calculated points shows that even in the worst case investigated (the cone) an accuracy of better than 10 per cent is reached at hypersonic similarity parameters of the order of 2.6. This indicates that the Newtonian concept is more than of academic interest; ballistic missiles and satellites now in planning may well fly for parts of their trajectories under conditions to which this theory applies.

Grimminger (4) used the Newtonian concepts to calculate the normal and chord-wise forces on bodies of revolution at an angle of attack. However, he did not concern himself with the lift and drag, i.e., forces perpendicular and parallel to the direction of relative motion. In hypersonic flow the drag force is large in relation to the lift. Moreover, the flight velocity is very large, the density is very low, etc., leading, in general, to a relative magnitude of forces, moments and accelerations that may be different from what is usually encountered. This necessitates a thorough review of many of the commonly made approximations. In particular, the problem of dynamic stability is taken up in this paper. Several typical bodies are studied to show the controlling elements of the stability problem. It is true that some important aspects of the analysis are lost in the oversimplifications of the Newtonian concept of flow; for instance, the theory does not yield any information on the dependence of the force coefficients on the Mach number of flight, but only their value as the Mach number approaches infinity. Nevertheless, the results are indicative of the character of the problem.

The starting point for the calculations is Equation [1] which expresses the Newtonian hypotheses

$$C_p = \frac{p - p_0}{\frac{1}{2}\rho V_0^2} = 2\cos^2(\bar{V}, \bar{n}) \quad [1]$$

This statement is valid only for the surfaces which "see" the flow. The approximation does not predict anything concerning the surfaces facing away from the flow; there, it must be assumed, the flow is completely separated and the local pressure p is equal to the free stream pressure p_0. This means that the local pressure coefficient in regions of expansion-flow is zero and Equation [1], integrated over the forward-facing surface, gives the total pressure force acting on the body.

The Aerodynamic Coefficients

A typical body of revolution is shown in Fig. 1 referred to its Cartesian axes. The origin of the coordinate system is at the nose, x is defined positive downstream, and y and z follow by the right-hand rule. The surface of the body is described in cylindrical coordinates r, x and β, where the angle is measured counterclockwise from the y-axis. The angle θ is used to denote the local slope of the surface with reference to the Ox-axis.

Let c denote the distance from the origin along the x axis to the center of gravity of the body. If the body is pitching at a rate q about the y-axis through the center of gravity, then the velocity induced over the surface of the body is

$$\bar{v} = \bar{q} \times \bar{R} \quad [2]$$

where

$$\bar{R} = (x - c)\bar{i} + r\bar{k}$$

$$\bar{q} = q\bar{j}$$

The free stream velocity V_0 has a component U along the

Received Aug. 27, 1956.

[1] Assistant Professor, Department of Engineering.

[2] Numbers in parentheses indicate References at end of paper.

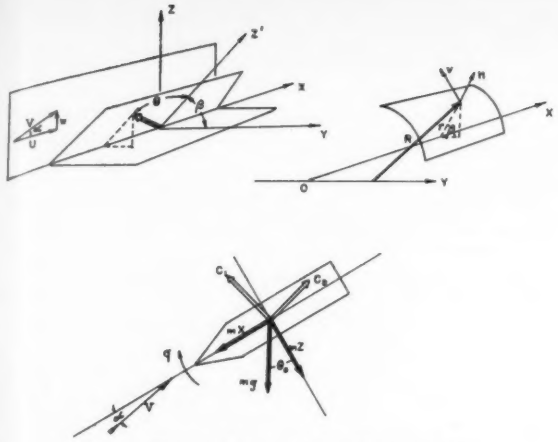

Fig. 1. Nomenclature and coordinate system

positive x-axis and w along the z-axis. The angle of attack α is given by

$$\alpha = \frac{w}{V_0} \quad \ldots\ldots [3]$$

Following Equation [1] it is now necessary to calculate the component of the total velocity normal to the surface. If $\bar{n}$ is the normal to the surface at any arbitrary point A, then from Fig. 1 the direction cosines of the normal $\bar{n}$, the free stream velocity V_0 and the pitching velocity v are as shown in Table 1.

Table 1 Direction cosines for body of revolution

	$\bar{n}$	V_0	v
x	$\sin\theta$	$\cos\alpha$	$-(r\sin\beta)/R$
y	$-\cos\theta\cos\beta$	0	0
z	$-\cos\theta\sin\beta$	$\sin\alpha$	$(x-c)/R$

The pressure coefficient is expanded in terms of the direction cosines of Table 1

$$V^2\cos^2(\bar{V},\bar{n}) = [V_0\cos(\bar{V}_0,\bar{n}) + v\cos(\bar{v},\bar{n})]^2 \ldots [4]$$

and

$$v = Rq = q\sqrt{(x-c)^2 + r^2\sin^2\beta}$$

If we use the direction cosines from Table 1, the pressure coefficient becomes

$$C_p = 2\cos^2\alpha\sin^2\theta -$$

$$\sin\beta\sin 2\theta[\sin 2\alpha + \frac{2q}{V_0}(r\tan\theta + x - c)\cos\alpha] +$$

$$2\sin^2\beta\cos^2\theta\,[\sin\alpha + \frac{q}{V_0}(r\tan\theta + x - c)]^2 \ldots [5]$$

For the particular case of axisymmetric bodies of revolution, the slope of the surface is independent of the meridian angle β. The body is uniquely determined by the co-planar coordinates r, x and θ. The expression [5] can therefore be substantially condensed by writing

$$C_p = 2\frac{r'^2}{1+r'^2}\cos\alpha -$$

$$2\frac{r'}{1+r'^2}\left[\sin 2\alpha + 2\frac{q}{V_0}(rr' + x - c)\cos\alpha\right]\sin\beta +$$

$$2\frac{1}{1+r'^2}\left[\sin\alpha + \frac{q}{V_0}(rr' + x - c)\right]^2\sin^2\beta \ldots [6]$$

The force coefficients of interest are C_n and C_c which act normal and parallel, respectively, to the x-axis. These coefficients will be arbitrarily referred to the maximum cross-sectional area of the body in question, which is $s = \pi r_B^2$. To obtain the force normal to the surface, the pressure coefficient times the element of surface ($ds = [r/\cos\theta]\,dr d\beta$) must be integrated with β and x. To decompose the total force into the normal and chord-wise components the appropriate direction cosines are used, so that

$$C_n = \frac{1}{2r_B}\int_0^l C_n' dx = -\frac{2}{\pi r_B^2}\int_0^l\int_{\beta_L}^{\beta_U} rC_p\sin\beta d\beta dx$$

$$C_c = \frac{1}{2r_B}\int_0^l C_c' dx = \frac{2}{\pi r_B^2}\int_0^l\int_{\beta_L}^{\beta_U} rr'C_p d\beta dx$$

The limits of integration on β are defined by the condition that $C_p = 0$, in accord with the assumptions of the Newtonian theory. However, the general solution for β_U and β_L from Equation [5] is algebraically very involved. It is therefore more convenient to consider the limits in particular cases separately. In the work that follows we shall only need the following solutions:

(a) The entire body surface "sees" the flow: ($\beta_U = +\pi/2$; $\beta_L = -\pi/2$)

$$C_n' = 4\frac{r}{r_B}\frac{r'}{1+r'^2}\left[\sin 2\alpha + 2\frac{q}{V_0}(rr' + x - c)\cos\alpha\right] \ldots [7a]$$

$$C_c' = 4\frac{r}{r_B}\frac{r'}{1+r'^2}\times$$
$$\left[2r'^2\cos^2\alpha + \left\{\sin\alpha + \frac{q}{V_0}(rr' + x - c)\right\}^2\right] \ldots [8a]$$

(b) Only one half of the cross-sectional surface sees the flows: ($\beta_U = 0$; $\beta_L = -\pi/2$)

$$C_n' = \frac{8}{\pi}\frac{r}{r_B}\left\{\frac{r'^2}{1+r'^2}\cos^2\alpha + \right.$$
$$\frac{r'}{1+r'^2}\frac{\pi}{4}\left[\sin 2\alpha + 2\frac{q}{V_0}(rr' + x - c)\cos\alpha\right] +$$
$$\left.\frac{2}{3}\frac{1}{1+r'^2}\left[\sin\alpha + \frac{q}{V_0}(rr' + x - c)\right]^2\right\} \ldots [7b]$$

$$C_c' = 4\frac{r}{r_B}\frac{r'}{1+r'^2}\times$$
$$\left\{r'^2\cos^2\alpha + \frac{2}{\pi}r'\left[\sin 2\alpha + 2\frac{q}{V_0}(rr' + x - c)\cos\alpha\right] + \right.$$
$$\left.\frac{1}{2}\left[\sin\alpha + \frac{q}{V_0}(rr' + x - c)\right]^2\right\} \ldots [8b]$$

In addition to the force coefficients, the pitching moment coefficient is of importance. Considering the component of the force in the xz-plane normal to the radius R from the y-axis passing through the center of gravity to the element of surface, one derives exactly as before

$$C_m = \frac{M}{\frac{1}{2}\rho V_0^2 sl} = -\frac{1}{2r_B l}\int_0^l C_n'(rr' + x - c)dx \ldots [9]$$

The Cone[3]

Before proceeding, it is necessary to specify the nature of the surface. One of the simplest nontrivial bodies of revolu-

[3] Since this article was presented for publication, Ref. (9) has been released. It contains a discussion of the stability derivatives derived by Newtonian theory and an interesting comparison of that approximation with potential theory.

tion which has direct application to hypersonic missiles and projectiles is the circular cone. It is described by the relation

$$r = x \tan \theta$$

$$r' = \tan \theta = \text{constant} \quad \ldots\ldots [10]$$

Introducing this into Equations [7a, 8a], the integration results in the following expressions for the three basic coefficients

$$C_n = \cos^2 \theta \sin 2\alpha + 2\left(\frac{ql}{V_0}\right) \cos \alpha \left(\frac{2}{3} - \frac{c}{l} \cos^2 \theta\right) \quad \ldots\ldots [11a]$$

$$C_c = 2 \sin^2 \theta + \sin^2 \alpha \,(1 - 3 \sin^2 \theta) + 2\left(\frac{ql}{V_0}\right) \sin \alpha \left(\frac{2}{3} - \frac{c}{l} \cos^2 \theta\right) + \sin \alpha \left(\frac{ql}{V_0}\right)^2 \sec^2 \theta \times \left[\left(\frac{2}{3} - \frac{c}{l} \cos^2 \theta\right)^2 + \frac{1}{18}\right] \ldots [11b]$$

$$C_m = -\sin 2\alpha \left(\frac{2}{3} - \frac{c}{l} \cos^2 \theta\right) - \cos \alpha \left(\frac{ql}{V_0}\right)\left[\left(\frac{2}{3} - \frac{c}{l} \cos^2 \theta\right)^2 + \frac{1}{18}\right] \sec^2 \theta \ldots [11c]$$

Because of the limits implied in Equations [7a, 8a] these relations are restricted to the case where the entire cone surface has a component normal to the flow, i.e., to the case where $\theta \geq \alpha$. The first term of each of the above expressions is identical to the result of (4); the remaining two terms are due to the pitching velocity. The factor $(\frac{2}{3} - (c/l) \cos^2 \theta)$ is associated with the location of the center of pressure in reference to the center of gravity (about which the moments are taken): For instance, when q vanishes

$$\frac{\bar{x}}{l} = \frac{2}{3} \sec^2 \theta = \frac{2}{3}(1 + \tan^2 \theta) \quad \ldots\ldots [12]$$

which exemplifies the contribution of the chord-wise force to the moment. Only for a vanishingly small cone angle is the center of pressure located at $\frac{2}{3}$ of the chord. As the cone angle increases, the center of pressure moves rearward. Its location has an important influence on the stability of the body. It can also be noticed how the center of pressure is shifted when the pitching velocity is different from zero.

Likewise, it is not possible to disregard the contribution of the chord-wise force to the lift and drag. These are given by

$$C_L = C_n \cos \alpha - C_c \sin \alpha$$

$$C_D = C_n \sin \alpha + C_c \cos \alpha$$

The initial slope of the lift coefficient curves is

$$\left.\frac{dC_L}{d\alpha}\right|_{\alpha \to 0} = 2 \cos 2\theta \quad \ldots\ldots [13]$$

For slender cones this reduces to the results of (4), but for finite semivertex angles it differs substantially from them.

The variation of C_L and C_D is shown in Fig. 2 for $\theta = 30$ deg. The curves have a shape which is surprisingly similar to their low speed shape, including a maximum lift coefficient, but this is entirely coincidental. It is also interesting to note that there exists an angle for which the lift to drag ratio is optimum.

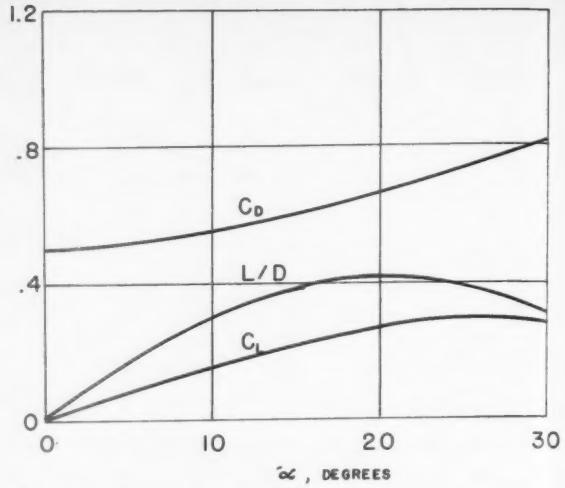

Fig. 2 Aerodynamic force coefficients for a cone with a semivertex angle of 30 deg

The Stability Conditions

A balance of forces and moments on a body in flight results in a quartic whose coefficients govern the dynamic stability of the motion. Detailed discussions of the problem are found in (6 and 7). The motion will be stable when

$$\left.\begin{array}{l} b, c, d, e > 0 \\ bcd - d^2 + b^2 e > 0 \end{array}\right\} \quad \ldots\ldots [14]$$

The relation between these coefficients and the aerodynamic derivatives appears in Table 2.

In writing out the stability problem for Newtonian flow, several terms deserve attention. The quasi-steady accelerations $Z_{\dot{w}}$ and $M_{\dot{w}}$ do not appear because there is no interaction within the flow field. The derivatives with respect to the flight velocity C_{LV}, C_{DV} and C_{mV} are zero, which is inherent in the Newtonian approximation. However, the usual

Table 2 The coefficients of the stability quartic

$b/K = \sum$	+		C_D
	−	$\frac{1}{2}P$	C_{mq}
	+	$\frac{1}{2}$	$(C_{L\alpha} + C_D)$
$c/K^2 = \sum$	−	$\frac{1}{2}P$	$C_D C_{mq}$
	−	$\frac{1}{2}PQ$	$C_{m\alpha}$
	+	$\frac{1}{4}P$	$C_{m\alpha} C_{Lq}$
	−	$\frac{1}{2}P$	$C_{mq}(C_{L\alpha} + C_D)$
	−	$\frac{1}{2}$	$C_L(C_{D\alpha} - C_L)$
	+	$\frac{1}{2}$	$C_D(C_{L\alpha} + C_D)$
$d/K^3 = \sum$	+	$(g/VK)\frac{1}{2}PQ$	$C_{m\alpha} \sin \theta_0$
	−	$\frac{1}{4}P$	$C_D C_{mq}(C_{L\alpha} + C_D)$
	−	$\frac{1}{4}P$	$C_L C_{Dq} C_{m\alpha}$
	−	$\frac{1}{2}PQ$	$C_D C_{m\alpha}$
	+	$\frac{1}{4}P$	$C_D C_{Lq} C_{m\alpha}$
	+	$\frac{1}{4}P$	$C_L C_{mq}(C_{D\alpha} - C_L)$
$e/K^4 = \sum$	−	$(g/VK)\frac{1}{2}PQ$	$C_{m\alpha} C_L \cos \theta_0$
	+	$(g/VK)\frac{1}{2}PQ$	$C_{m\alpha} C_D \sin \theta_0$

assumption that the forces are not dependent on the accelerations is not so obvious for hypersonic flow. If these accelerations are not too large, then their effect can be viewed in terms of a "virtual mass" of the accelerating body, and their neglect may continue to be justified. The virtual mass is proportional to the mass of the air displaced and is inversely proportional to the flight velocity. Because of the probable magnitude of these terms, the virtual mass is very small and it is reasonable to suppose that its effect on the aerodynamic force structure will be negligible. This will be assumed in what follows.

The terms of the stability coefficients contain factors dependent on the flight condition and the aerodynamic derivatives. The latter are functions of the geometry of the body and, for reasonable geometries, their magnitudes are of the order of one, as numerical examples discussed subsequently demonstrate. The former, which are

$$\left.\begin{aligned} K &= \frac{(\pi r_B^2)\rho V_0}{m} \\ P &= \frac{ml^2}{B} \\ Q &= \frac{m}{\rho(\pi r_B^2)l} \end{aligned}\right\} \quad \ldots\ldots\ldots\ldots\ldots\ldots [15]$$

have magnitudes of the order of 10^{-1}, 1 and 10^5, respectively, for typical hypersonic flight conditions.

Consequently, if $C_{m\alpha}$ is not zero (that is, away from the point of neutral static stability) the important terms of the coefficients of the stability quartic are

$$\left.\begin{aligned} b &= \frac{1}{2}K[-PC_{mq} + C_{L\alpha} + 3C_D] = KA \\ c &= -\frac{1}{2}K^2PQC_{m\alpha} = K^2QC \\ d &= \frac{1}{2}K^3PQC_{m\alpha}\left[\frac{g}{V_0K}\sin\theta_0 - C_D\right] = \\ &\qquad - K^2QC\left[\frac{g}{V_0K}\sin\theta_0 - C_D\right] \\ e &= \frac{1}{2}K^4PQC_{m\alpha}\frac{g}{V_0K}[C_D\sin\theta_0 - C_L\cos\theta_0] = \\ &\qquad - K^4QC[C_D\sin\theta_0 - C_L\cos\theta_0]\frac{g}{V_0K} \end{aligned}\right\} \quad \ldots [16]$$

The factors A and C are of order one. In an equilibrium unaccelerated flight for which the static force balance (assuming small angles of attack) gives

$$\left.\begin{aligned} D &= -mg\sin\theta_0 \\ L &= -mg\cos\theta_0 \end{aligned}\right\} \quad \ldots\ldots\ldots\ldots\ldots [17]$$

Routh's discriminant can then be written

$$A - \frac{3}{2}C_D + \frac{A^2}{C}\frac{1}{3}\frac{C_D(1 - \cot^2\theta_0)}{Q} \quad \ldots\ldots\ldots [18]$$

The third term is very small as long as $\cot^2\theta_0$ is reasonably small. This is so from the vertical dive ($\theta_0 = 90$ deg) to at least a 45 deg glide, for which case that term is actually zero. Equation [18] reduces then to the simple condition

$$\frac{1}{2}[C_{L\alpha} - PC_{mq}] > 0 \quad \ldots\ldots\ldots\ldots\ldots [19]$$

which is valid for an unaccelerated, reasonably steep dive of a statically stable body of revolution.

The Stability of a Cone

Under the assumption of vanishing pitching and small angle of attack, so that second order terms in α_0 can be neglected, the stability derivatives for the cone are

$$\left.\begin{aligned} C_D &= 2\sin^2\theta \\ C_L &= 2\alpha_0\cos 2\theta = \alpha_0 C_{L\alpha} = \frac{1}{3}C_{D\alpha} \\ C_m &= -2\alpha_0\left(\frac{2}{3} - \frac{c}{l}\cos^2\theta\right) = \\ &\qquad \alpha_0 C_{m\alpha} = -\alpha_0 C_{Lq} = -\frac{1}{2}C_{Dq} \\ C_{mq} &= -2\sec^2\theta\left[\left(\frac{2}{3} - \frac{c}{l}\cos^2\theta\right)^2 + \frac{1}{18}\right] \end{aligned}\right\} \quad \ldots [20]$$

A conical body is capable of steady equilibrium flight only at zero angle of attack, since only then the steady state pitching moment is zero. The static stability margin is

$$\frac{C_{m\alpha}}{C_{n\alpha}} = \frac{C_{m\alpha}}{C_{L\alpha} + C_D} = -(1 + \tan^2\theta)\left(\frac{2}{3} - \frac{c}{l}\cos^2\theta\right) \quad \ldots [21]$$

and it is negative only if the center of gravity is located forward of the center of pressure.

If the cone is homogeneous, the center of gravity is located $\frac{3}{4}$ of its length from the apex. It is statically unstable unless its half-angle is larger than 19.5 deg. For example, a 20 deg half angle will provide approximately $\frac{1}{2}$ per cent static stability margin. The dynamic stability coefficients for such a cone are subject to the assumptions in Equation [19] because, although $C_{m\alpha}$ is small, $QC_{m\alpha} \simeq (10^5 \times 10^{-2})$ is still very large in comparison with the other items.

The cone in this example is stable provided it is diving ($\sin\theta_0$ is negative). It can be said quite generally that [19] will be satisfied when the lift-curve slope is positive because $C_{m\alpha}$ is always negative. However, when the cone angle is larger than 45 deg $C_{m\alpha}$ becomes negative and then it is possible to stipulate a condition for which the dynamic stability will break down. For a homogeneous cone this will occur in the limit as it approaches a plate (θ approaches 90 deg), but it may occur sooner if the mass distribution in the body is such as to make P sufficiently small.

At the neutral stability point $C_{m\alpha} = 0$ and all terms containing Q vanish. The coefficient e is zero and there will exist one neutrally damped mode of oscillations. Equation [19] does not apply and account must be made of all the remaining terms. Inspection of Table 2 shows that they are all positive and Routh's discriminant is identically positive. If one now considers this case as the limit when $QC_{m\alpha}$ is small (so that the coefficient e is positive approaching zero), the cone remains dynamically stable to the limit of its static stability provided Equation [19] is satisfied.

The Hemispherical Forebody

A hemisphere is an interesting example of a blunt body. Dealing with it, however, is algebraically very involved, because the limit of the surface which "sees" the flow is a function of both x and q. Instead, let us assume a spherical sector of half-angle γ_0 such that $(\pi/2 - \gamma_0)$ is smaller than the net induced angle of attack due to inclination to the flow

and pitching. Then $\beta_U = \pi/2$ and Equations [7a, 8a] are applicable with the substitutions

$$r = \rho \sin \gamma$$

$$r_B = \rho \sin \gamma_0$$

$$x = \rho(1 - \cos \gamma)$$

One obtains

$$C_n = \frac{1}{2}\left[\sin 2\alpha + \left(\frac{q\rho}{V_0}\right)\left(1 - \frac{c}{\rho}\right)\cos \alpha\right] \sin^2 \gamma_0$$

$$C_c = \frac{1}{2}\left[\sin \alpha + \left(\frac{q\rho}{V_0}\right)\left(1 - \frac{c}{\rho}\right)\right]^2 \sin^2 \gamma_0 + \cos^2 \alpha\,(1 + \cos^2 \gamma_0)$$

It follows,[4] for a vanishing α and q

$$\left.\begin{aligned} C_L &= C_m = C_{D\alpha} = C_{Dq} = 0 \\ C_D &= 1 + \cos^2 \gamma_0 \\ C_{L\alpha} &= -2 \cos^2 \gamma_0 \\ C_{m\alpha} &= -C_{Lq} = -\left(1 - \frac{c}{\rho}\right) \sin^2 \gamma_0 \\ C_{mq} &= -\left(1 - \frac{c}{\rho}\right)^2 \sin^2 \gamma_0 \end{aligned}\right\} \quad \ldots [22]$$

The spherical sector behaves similarly to the cone. It has only one equilibrium flight condition for which the moment vanishes which is statically stable if $c/\rho < 1$ and $\sin \theta_0 = -1$, that is, a vertical dive. However, there is this notable difference: The lift curve slope is negative for all values of the sector half angle. Equation [19] requires that

$$P\left(1 - \frac{c}{\rho}\right)^2 > 2 \cot^2 \gamma_0 \quad \ldots [23]$$

which indicates (for a given position of the center of gravity) that large half-angle sectors will be dynamically stable while "slender" sectors will not.

This constitutes a good example of the importance of the drag force in Newtonian flow.

The Cone-Cylinder

Suppose that a conical nose is followed by a cylindrical afterbody. The force coefficients for the cylinder can be readily obtained from the Equations [7b, 8b] by inserting characteristic values

$$r = r_B = \text{const}, \qquad r' = 0, \qquad \beta_U = 0$$

The chord-wise force is identically zero. The normal force is

$$C_n = \frac{8}{3\pi}\frac{l_1}{r_B}\sin^2 \alpha + \frac{8}{9\pi}\frac{l_1}{r_B}\left(\frac{ql_1}{V_0}\right)^2\left(\frac{(l_1 - c_1)^3 + c_1^3}{l_1^3}\right) + \frac{8}{3\pi}\frac{l_1}{r_B}\left(\frac{ql_1}{V_0}\right)\left(1 - 2\frac{c_1}{l_1}\right)\sin \alpha$$

and the other force and moment coefficients, for vanishing pitching, are[5]

[4] The moment coefficient is referred to the radius of the generating sphere ρ.

[5] The moment coefficient is referred to the length of the cylinder l_1.

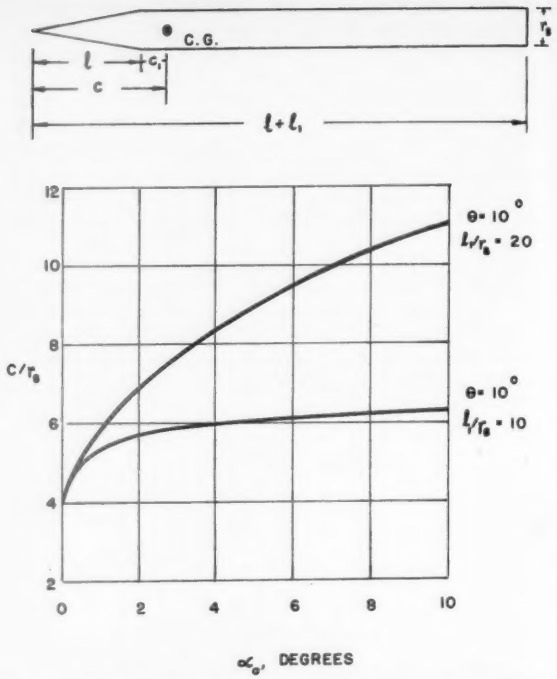

Fig. 3 Equilibrium angle of attack for a gliding cone-cylinder as a function of the location of the center of gravity

$$\left.\begin{aligned} C_L &= \frac{8}{3\pi}\frac{l_1}{r_B}\sin^2 \alpha \cos \alpha \\ C_D &= \frac{8}{3\pi}\frac{l_1}{r_B}\sin^3 \alpha \\ C_m &= -\frac{4}{3\pi}\sin^2 \alpha\left(\frac{l_1}{r_B} - 2\frac{c_1}{r_B}\right) \end{aligned}\right\} \quad \ldots [24]$$

It is evident that the dynamic stability of the cone-cylinder is identical to that of the cone itself (for a given location of the center of gravity) when the angle of attack α_0 equals zero. All of the coefficients and their derivatives for the cylinder vanish and the afterbody contributes no force or moment regardless of its size relative to the nose.

However, if the angle of attack is allowed to increase, the contribution of the cylindrical afterbody grows more rapidly than that of the cone. There will be a definite angle of attack where the moments about the center of gravity cancel, provided the c.g. lies between the c.p. of the cone and that of the body. This will occur when

$$\frac{M_{nose}}{M_{cyl}} = \frac{C_{m\ nose}}{C_{m\ cyl}}\frac{l}{l_1} = \frac{3\pi}{2}\cot \alpha_0\left[\frac{\frac{2}{3}\cot \theta - \frac{c}{r_B}\cos^2 \theta}{\frac{l_1}{r_B}\left(\frac{l_1}{r_B} - 2\frac{c}{r_B} + 2\cot \theta\right)}\right] = -1 \quad \ldots [25]$$

At $\alpha_0 = 0$, such a cone-cylinder is statically unstable. However, as α_0 increases, its static stability increases until the neutral point is reached when

$$\frac{C_{m\alpha\ nose}}{C_{m\alpha\ cyl}}\frac{l}{l_1} = \frac{3\pi}{2}\cot 2\alpha_0\left[\frac{\frac{2}{3}\cot \theta - \frac{c}{r_B}\cos^2 \theta}{\frac{l_1}{r_B}\left(\frac{l_1}{r_B} - 2\frac{c}{r_B} + 2\cot \theta\right)}\right] = -1 \quad \ldots [26]$$

It is evident that, whatever the geometry of the cone-cylinder, Equation [26] is satisfied for a smaller α_0 than Equation [25]. Therefore, there will always exist an angle of attack α_0 where statically stable trimmed-out flight is possible. Its magnitude will vary with the location of the center of gravity. Fig. 3 shows this variation for a sample cone angle and two cylinder fineness ratios. The above specified conditions imply of course that there is finite lift. Therefore a direct dive is not possible. The body will follow a path on which an equilibrium between the lift force and the opposing component of the gravitational acceleration is established.

Assuming now that the trimmed-out angle of attack is small so that only first-order terms are retained, the coefficients for the cylindrical afterbody become

$$\left.\begin{aligned} C_L &= C_D = C_{n\alpha} = C_{Dq} = 0 \\ C_{L\alpha} &= \frac{16}{3\pi}\frac{l_1}{r_B}\alpha_0 \\ C_{m\alpha} &= -C_{Lq} = -\frac{8}{3\pi}\alpha_0\left[\frac{l_1}{r_B} - 2\frac{c}{r_B} + 2\cot\theta\right] \\ C_{mq} &= \frac{16}{9\pi}\alpha_0\left[\frac{l_1}{r_B} - 3\left(\frac{c}{r_B} - \cot\theta\right) + 3\frac{r_B}{l_1}\left(\frac{c}{r_B} - \cot\theta\right)^2\right] \end{aligned}\right\} \quad \ldots\ldots [27]$$

Note that the approximation introduces an inconsistency whereby the moment contributions due to the nose and the afterbody do not cancel (as they should from the exact relations). However, this is inconsequential since all stability terms containing C_m are much smaller than the others.

For a cylinder dimension $l_1/r_B = 10$, a cone half angle $\theta = 10$ deg and a trimmed-out angle $\alpha_0 = 3$ deg, from Equation [25], the position of the center of gravity must be $c/r_B = 5.790$. Note that the center of gravity is somewhat downstream of the conical nose. A sketch to scale of the assumed configuration appears in Fig. 3.

In this example the flight-path angle θ_0 as computed from Equation [17], i.e., for an unaccelerated equilibrium flight, is 32.3 deg. It is surprising to find how a relatively small shift in the location of the center of gravity alters the equilibrium trajectory.

Thus, two flight-path angles for equilibrium are possible for this configuration. The first one occurs if the center of gravity of the body is forward of the center of pressure of the nose, and it is still the vertical dive. The dynamic stability is then controlled by the geometry of the nose only. The second equilibrium condition occurs when the center of gravity is aft of the center of pressure of the nose. Corresponding to it there is a unique angle of attack and a unique value of the lift force; consequently the body will follow a determined trajectory. Since Equation [19] remains applicable, it is easy to see that the second trajectory is dynamically stable (in fact, more so than the first one because C_{mq} is much larger).[6]

A very interesting peculiarity of the Newtonian flow force-field is notable for the nose-afterbody configuration. The second equilibrium condition (at an angle of attack) disappears if instead of a cylinder a truncated cone is used for the afterbody. This is because, provided that the total angle of attack does not exceed the half-cone angle of the afterbody, the moment contributions of both the fore and aft sections grow linearly with angle of attack and do not cancel (except at $\alpha_0 = 0$). Only when the upper half of the afterbody becomes submerged in the Newtonian wake (as is the case with the cylinder), the rates of change of the contribution of the afterbody and the forebody to the moment with changing angle of attack differ, and it becomes possible to establish an angle of attack different from zero at which they cancel.

The behavior described above will be also exhibited by bodies with other nose geometries such as spherical sectors or ogives. Also, horizontal control surfaces (flat plates) will act in a manner similar to that of the cylindrical afterbody.

[6] In the calculations, account must be made for the proper reference-length which enter the definitions of moment coefficients.

Simplification of the Stability Problem

Equation [19] permits the evaluation of the dynamic stability of most probable (axisymmetric) airframe configurations under the conditions stated.

Conversely, a common approximation which consists in factoring the stability quartic into two quadratics (representing the phungoid and short-period oscillations, respectively) is not generally valid in hypersonic flow. This approximation can be made only if (7)

$$b \gg \frac{d}{c} - \frac{be}{c^2}$$

$$c^2 \gg e + db - \frac{b^2 e}{c}$$

Using Equations [16], assuming unaccelerated flight and a large value of Q, the inequality condition reverts to the condition that Routh's discriminant be very positive (see Equation [19]). This may be true for a slender cone or cone-afterbody configuration, but it is not satisfied for any reasonable spherical sector. For instance, for a homogeneous hemisphere it would be necessary to agree that $^{45}/_{64} \gg 0$ which is not a very good approximation.

Acknowledgment

In closing, the author wishes to express his gratitude to E. V. Laitone who drew his attention to this problem and contributed many stimulating remarks and discussions.

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Thermal Stresses in an Infinite, Hollow Case-Bonded Cylinder

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Formulas are derived in this paper relating thermal stresses to the temperature distribution in an infinitely long, hollow cylinder, bonded at its outer surface to a cylindrical shell, when the temperature of the outer shell is suddenly changed. Approximate relations are then obtained for the case of a thermally thin, but elastically thick shell corresponding to the practical configuration of a plastic propellant "grain," case-bonded to a metallic shell. It is argued that, for the heat problem considered, the stresses will be monotonic functions of time, and hence the maximum stresses will occur either initially or finally. Since the temperature distribution can be simply described in these limiting cases, simple formulas for the stresses can be written down. The solutions given in the paper for an infinite cylinder may be expected to approximate the stresses in a long but finite cylinder in regions where end effects can be neglected.

Nomenclature

a = inner radius (unstrained) of grain
b = radius of grain-shell interface
c = outer radius of shell
d = constant of integration
E = Young's Modulus
e = trace of strain tensor
$\mathbf{e}_t$ = unit vector in the tangential (as opposed to radial or axial) direction
F_z = axial force (assumed to vanish)
I = unit tensor
$\mathbf{k}$ = unit vector in the axial (z) direction
k = $\alpha E/(1-\sigma)$, stress parameter
M = c/b, m = a/b } radius (diameter) ratios of shell-grain configuration
T = temperature variation from initial state
$\bar{T}$ = temperature integral
t = trace of stress tensor
$\mathbf{u}$ = displacement vector
$u_{\rho,z}$ = radial, axial components of displacement
z = axial coordinate
α = linear coefficient of thermal expansion
δ,ϵ,μ = interface parameters
ϵ = strain tensor
θ = temperature parameter
λ = $E/(1+\sigma)$, stress parameter
ρ = radial coordinate
$\nabla\rho$ = unit vector in radial direction
σ = Poisson ratio
$\sigma_{\rho,t,z}$ = radial, tangential, axial stresses
τ = stress tensor (components $\tau_{zz} = \sigma_z$, $\tau_{\rho z}$, etc.)
$\tau_{1,0}$ = mean temperatures in grain (1) or shell (0)
ϕ = axial strain (used as a parameter)

The subscripts 1, 0 refer to the grain and shell, respectively.

Introduction

THE problem considered here concerns the stresses which arise in a concentric annular cylinder of infinite length, bonded at its outer surface to a cylindrical shell, when the temperature of the outer shell is suddenly changed. The inner surface of the cylinder is supposed to be thermally insulated. It is assumed that the system is free of stress in its initial state;[2] such would be the case if enough time were allowed (before the temperature change occurs) for viscous flow to relieve any stresses which might have been present in the materials. However, viscous flow will not be considered in the problem. Thus, when thermal equilibrium has again set in, the system will in general be under a residual stress.

In treating the stress problem, the quasi-steady-state approximation will be used: The stresses correspond to the instantaneous temperature distribution. This approximation omits the possibility of the propagation of elastic waves, but the error involved is not important for the situation considered. The time then enters the stress equations only as a parameter in the temperature distribution; this dependence will be suppressed in the analysis.

In the discussion to follow, we shall be mainly concerned with the stress relations, and only incidentally with the temperature distribution. So long as the displacements due to the thermal stresses are not large, the temperature distribution can be obtained by solving a problem completely separate from the stress problem. We assume that the heat transfer problem has been solved, and consider the temperatures to be known at the instant of time for which the stress equations are written. From the description of the problem given above, it is evident that the temperature field at a given time will depend only on the radial distance from the axis of symmetry of the system.

The practical problem, for which this investigation was begun, is that of thermal stresses in a hollow, case-bonded, rocket propellant casting. The inner cylinder is then constructed of a plastic material, and the outer cylinder is metallic. With this configuration in mind, we shall find it convenient to refer to the inner cylinder as the "grain" and the outer material as the "shell." However, except for the final sections, in which physical assumptions pertinent to this case are made, the analysis is quite general. The solutions for the infinite cylinder may be expected to approximate the stresses in a long, but finite, cylinder in regions not too near the ends.

Equilibrium Equations

Because of the cylindrical symmetry of the configuration, the (vector) displacement $\mathbf{u}$ of the material from its initial position can, at any point, have only an axial (z) or radial (ρ) component, and can depend only on the coordinates z and ρ (see Fig. 1). Hence the rotation vector $\nabla\times\mathbf{u}$ will have only a "tangential" component (be everywhere tangent to the circles of constant ρ), and so can be represented as

$$\nabla\times\mathbf{u} = -\mathbf{k}\times\nabla f(\rho, z) = \nabla\times\mathbf{k}f$$

where $\mathbf{k}$ is a unit vector in the z direction. It follows by in-

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[1] Physicist, Liquid Engine Division.

[2] Gravitational or accelerational stresses are ignored.

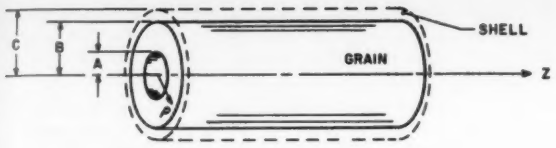

Fig. 1 Grain and shell configuration

tegration that **u** differs from **k**f only by the gradient of a scalar

$$\mathbf{u} = \mathbf{k}, f(\rho, z) + \nabla g(\rho, z) \quad \ldots\ldots [1]$$

For an infinite cylinder, the radial component of displacement $u_\rho = \partial g/\partial \rho$ must be independent of z

$$\frac{\partial u_\rho}{\partial z} = \frac{\partial^2 g}{\partial z \partial \rho} = 0$$

whence

$$g(\rho, z) = v(\rho) + w(z), \text{ say} \quad \ldots\ldots [2]$$

and the axial component of displacement u_z can (by a suitable choice of origin) be taken proportional to z. Thus, from Equations [1, 2]

$$\left.\begin{aligned} u_z &= f(\rho, z) + w'(z) = z\phi(\rho) \\ \text{or} \\ f(\rho, z) &= z\phi(\rho) - w'(z) \end{aligned}\right\} \quad \ldots\ldots [3]$$

where $w' = dw/dz$. Combining Equations [1, 2 and 3], we obtain

$$\mathbf{u} = \nabla v(\rho) + \mathbf{k}z\phi(\rho) \quad \ldots\ldots [4]$$

from which the terms involving $w(z)$ have disappeared.

The strain tensor (in dyadic form) is given by[3]

$$\epsilon = \frac{1}{2}[\nabla \mathbf{u} + (\nabla \mathbf{u})'] \quad \ldots\ldots [5]$$

in terms of the gradient of **u**

$$\nabla \mathbf{u} = \nabla\nabla v + (\mathbf{k}\phi + z\,\nabla\phi)\mathbf{k}$$

and the transposed tensor $(\nabla u)'$; thus

$$\epsilon = \nabla\nabla v + \mathbf{k}\mathbf{k}\phi + \frac{1}{2}z(\mathbf{k}\,\nabla\phi + \nabla\phi\mathbf{k}) \quad \ldots\ldots [6]$$

The stress-strain relation for an isotropic material under thermal stress is given by

$$\epsilon = \frac{1+\sigma}{E}\tau - \frac{\sigma}{E}tI + \alpha T I \quad \ldots\ldots [7]$$

where σ, E and α are the Poisson ratio, Young's modulus, and linear coefficient of thermal expansion of the material; T is the temperature change from the initial state; and the symbols τ, t and I here represent the strain tensor, its trace (diagonal sum) $t = \Sigma_i \tau_{ii}$, and the unit tensor. By solving Equation [7] for the stress tensor, one obtains

$$\tau = \lambda\left\{\epsilon + \left[\frac{\sigma e - \alpha(1+\sigma)T}{1-2\sigma}\right] I\right\} \quad \ldots\ldots [8]$$

where

$$e = tr(\epsilon) = \sum_i \epsilon_{ii} \qquad \lambda = \frac{E}{1+\sigma}$$

[3] For the definitions given here of the strain and stress tensors (Equations [5, 7]), the equilibrium Equation [9], and the boundary conditions, we may cite Timoshenko and Goodier (Reference 1) as standard reference. Some translation may be needed, however, since this book uses the long notation.

It is supposed in the following that α, σ and E (and hence λ) are independent of temperature, or have been given appropriate mean values.

For the material to be in equilibrium under stress, the divergence of the stress tensor must vanish at each point. From Equations [8, 5], this condition can be written

$$\nabla\cdot\tau = \lambda\left\{\frac{1}{2}\nabla^2\mathbf{u} + \frac{1}{2}\nabla(\nabla\cdot\mathbf{u}) + \nabla\left[\frac{\sigma e - \alpha(1+\sigma)T}{1-2\sigma}\right]\right\} = 0 \quad \ldots\ldots [9]$$

The curl of this relation is

$$\nabla\times(\nabla\cdot\tau) = \frac{1}{2}\lambda\nabla^2(\nabla\times\mathbf{u}) = \frac{1}{2}\lambda\mathbf{k}\times\nabla(\nabla^2 z\phi)$$

according to Equation [4], and this must also vanish. Thus $\nabla^2 z\phi(\rho) = z\nabla^2\phi(\rho)$ must be a function of z alone, and therefore

$$\nabla^2\phi = 2K$$

must be a constant. Then, from Equation [4]

$$\nabla^2\mathbf{u} = \nabla^2(\nabla v) + \mathbf{k}(2Kz) = \nabla(\nabla^2 v + Kz^2) \quad \ldots\ldots [10]$$

and

$$\nabla(\nabla\cdot\mathbf{u}) = \nabla(\nabla^2 v) + \nabla\phi$$

and Equation [9] reduces to

$$\nabla\cdot\tau = \lambda\nabla\left\{\nabla^2 v + \frac{1}{2}(\phi + Kz^2) + \left[\frac{\sigma e - \alpha(1+\sigma)T}{1-2\sigma}\right]\right\} = 0$$

or

$$\nabla^2 v + \frac{1}{2}(\phi + Kz^2) + \left[\frac{\sigma e - \alpha(1+\sigma)T}{1-2\sigma}\right] = \text{const} \quad \ldots\ldots [11]$$

In Equation [11], v, ϕ and T depend only on ρ, while from Equation [6]

$$e = \nabla^2 v + \phi \quad \ldots\ldots [12]$$

Hence, Equation [11] cannot hold unless the term involving z vanishes, and this requires that $K = 0$ in Equation [10]. The solution for ϕ will then be of the form

$$\phi = \text{const} + C \ln \rho$$

If the inner diameter of the grain is zero, the $\ln \rho$ term in ϕ would make **u** infinite on the axis (see Equation [4]) unless $C = 0$. If the inner diameter is not zero, the boundary condition at $\rho = a$ is that the normal (radial) component of the stress tensor vanish there. The radial stress is given by $\tau_\rho = \nabla\rho\cdot\tau$ ($\nabla\rho$ being a unit vector in the radial direction), and the z component of this is $\tau_{\rho z} = \nabla\rho\cdot\tau\cdot\mathbf{k}$. According to Equations [8, 6], $\tau_{\rho z} = \lambda\epsilon_{\rho z} = \frac{1}{2}\lambda z\phi'(\rho)$, which becomes

$$\tau_{\rho z} = \tfrac{1}{2}\lambda z\phi'(a) = \tfrac{1}{2}\lambda z\frac{C}{a} \qquad \text{at } \rho = a$$

If this is to vanish, we must again have $C = 0$. By a similar argument, C must vanish for the shell. Thus ϕ must in any case be a constant.

It is convenient to choose the constant in Equation [11] so that Equations [11, 12] give

$$\nabla^2 v = \frac{k}{\lambda}[T + (1-2\sigma)\theta] - 2\sigma\phi \quad \ldots\ldots [13]$$

where

$$k = \frac{\alpha E}{1-\sigma}$$

and θ is another constant.

Expressions for the Displacements and Stresses

The first integral of Equation [13] can be written

$$v'(\rho) = \frac{k\rho}{2\lambda}[\bar{T} + (1 - 2\sigma)\theta] - \rho\sigma\phi + \frac{d}{\lambda\rho} \quad \ldots\ldots [14]$$

where

$$\bar{T} = \frac{2}{\rho^2}\int^{\rho} xT dx$$

and d is constant of integration. The lower limit of integration for the temperature integral $\bar{T}$ in Equation [14] may be taken to be $\rho = a$ for the grain and $\rho = c$ for the shell (so that $\bar{T} = 0$ at the free boundaries).

To compute the stresses, we need

$$\nabla\nabla v = \nabla\left(\varrho\frac{v'}{\rho}\right) = \frac{v'}{\rho}\nabla\varrho + \rho\left(\frac{v''}{\rho} - \frac{v'}{\rho^2}\right)\nabla\rho\nabla\rho$$

$$= (\nabla^2 v)\nabla\rho\nabla\rho + \frac{v'}{\rho}[\nabla\varrho - 2\nabla\rho\nabla\rho]$$

Here

$$\nabla\varrho = \nabla(\varrho + \mathbf{k}z) - \mathbf{kk} = I - \mathbf{kk}$$

The unit tensor is conveniently written as $I = \nabla\rho\nabla\rho + \mathbf{kk} + \mathbf{e}_t\mathbf{e}_t$, where $\mathbf{e}_t$ is a unit vector in the tangential direction at each point, so that

$$\nabla\varrho = \nabla\rho\nabla\rho + \mathbf{e}_t\mathbf{e}_t$$

With the aid of this relation and the expression for $\nabla\nabla v$, we find from Equation [6] that

$$\epsilon = \nabla\nabla v + \mathbf{kk}\phi$$

$$= \left\{\frac{k}{\lambda}[T + (1 - 2\sigma)\theta] - 2\sigma\phi\right\}\nabla\rho\nabla\rho + \left\{\frac{k}{2\lambda}[\bar{T} + (1 - 2\sigma)\theta] - \sigma\phi + \frac{d}{\lambda\rho^2}\right\}[\mathbf{e}_t\mathbf{e}_t - \nabla\rho\nabla\rho] + \mathbf{kk}\phi$$

$$e = \nabla^2 v + \phi = \frac{k}{\lambda}[T + (1 - 2\sigma)\theta] + (1 - 2\sigma)\phi$$

The stresses may next be found from Equation [8], and the displacements from Equation [4]. After some reduction, these become

$$\left.\begin{aligned}\sigma_\rho &\equiv \tau_{\rho\rho} = \frac{1}{2}k(\theta - \bar{T}) - \frac{d}{\rho^2}\\ \sigma_t &\equiv \tau_{tt} = \frac{1}{2}k(\theta + \bar{T} - 2T) + \frac{d}{\rho^2}\\ \sigma_z &\equiv \tau_{zz} = E(\phi - \alpha\theta) + k(\theta - T)\end{aligned}\right\} \quad \ldots\ldots [15]$$

$$\tau_{\rho t} = \tau_{\rho z} = \tau_{zt} = 0$$

$$\left.\begin{aligned}u_z &= z\phi\\ u_\rho &= \rho\phi - \frac{\rho}{\lambda}[E(\phi - \alpha\theta) + \sigma_\rho]\end{aligned}\right\} \quad \ldots\ldots [16]$$

where from above

$$\lambda = \frac{E}{1 + \sigma} \qquad k = \frac{\alpha E}{1 - \sigma}$$

Boundary Conditions

At the free surfaces $\rho = a, c$, the normal stress σ_ρ must vanish; while, at the interface $\rho = b$ between the shell and the grain, the normal stress must be continuous. Because the grain is bonded to the shell at $\rho = b$, the shell and grain displacements must also be equal there. One further condition is needed to specify the problem. We take this to be the requirement of dynamic equilibrium: No net force (due to the thermal stresses) is present which would tend to translate the system along the z-axis. Any other force components will automatically vanish because of the cylindrical symmetry.

For the case $a = 0$ (solid grain), the radial component of displacement must vanish on the axis, and hence $\rho\sigma_\rho$ must vanish at $\rho = 0$ by Equation [16].

The continuity of displacement and normal stress at the interface $\rho = b$ will hold, by Equation [16], if the constant ϕ is the same for both media, and if

$$\frac{E_1(\phi - \alpha_1\theta_1) + \sigma_\rho}{\lambda_1} = \frac{E_0(\phi - \alpha_0\theta_0) + \sigma_\rho}{\lambda_0} \quad \ldots\ldots [17]$$

at $\rho = b$. Here the subscripts 1, 0 will be supposed to refer to the grain and shell, respectively. Equation [15] shows that at $\rho = b$

$$\sigma_\rho = \frac{1}{2}k_1(\theta_1 - \bar{T}_1) - \frac{d_1}{b^2} = \frac{1}{2}k_0(\theta_0 - \bar{T}_0) - \frac{d_0}{b^2} \quad \ldots [18]$$

The vanishing of σ_ρ at $\rho = a$ and c will follow if

$$d_1 = \frac{1}{2}a^2k_1\theta_1 \qquad d_0 = \frac{1}{2}c^2k_0\theta_0 \quad \ldots\ldots [19]$$

since $\bar{T}_1$ and $\bar{T}_0$ are defined to vanish at the respective free boundaries. Provided θ_1 remains finite as $a \to 0$, the first of Equations [19] indicates that, for a solid grain, d_1 should be taken equal to zero. This choice for d_1 will make $\rho\sigma_\rho$ vanish in the grain as $\rho \to 0$, according to Equation [15], so that the case of a solid grain can be obtained from that of an annular grain by letting the inner diameter of the grain tend to zero, and need not be treated separately, provided θ_1 remains finite as $a \to 0$.

Finally, the condition for dynamic equilibrium may be written

$$\frac{1}{\pi}F_z = 2\int_a^c \sigma_z\rho d\rho = (b^2 - a^2)[E_1(\phi - \alpha_1\theta_1) + k_1\theta_1] - b^2k_1\bar{T}_1 + (c^2 - b^2)[E_0(\phi - \alpha_0\theta_0) + k_0\theta_0] + b^2k_0\bar{T}_0 = 0 \quad \ldots [20]$$

where

$$\left.\begin{aligned}b^2\bar{T}_1 &= 2\int_a^b \rho T d\rho = \rho^2\bar{T}\Big|_{\rho = b} \quad \text{for the grain}\\ b^2\bar{T}_0 &= 2\int_c^b \rho T d\rho = \rho^2\bar{T}\Big|_{\rho = b} \quad \text{for the shell}\end{aligned}\right\} \quad \ldots [21]$$

It will be noted that, as defined, $\bar{T}_0$ is negative when T is positive in the outer shell.

According to Equations [18, 19]

$$2b^2\sigma_\rho(b) = (b^2 - a^2)k_1\theta_1 - b^2k_1\bar{T}_1 = (b^2 - c^2)k_0\theta_0 - b^2k_0\bar{T}_0 \quad \ldots\ldots [22]$$

By using Equation [22] in Equation [20], one can therefore reduce Equation [20] to

$$(b^2 - a^2)E_1(\phi - \alpha_1\theta_1) = (b^2 - c^2)E_0(\phi - \alpha_0\theta_0) \quad \ldots [23]$$

It is convenient to write

$$a = mb \qquad c = Mb$$

and put

$$\bar{T}_1 = (1 - m^2)\tau_1 \qquad \bar{T}_0 = (1 - M^2)\tau_0 \quad \ldots\ldots [24]$$

so that τ_1 and τ_0 give the average temperature in the grain and shell, respectively. With these definitions, the boundary relations in Equations [17, 22, 23] become

$$2\sigma_\rho = (1 - m^2)k_1(\theta_1 - \tau_1) = (1 - M^2)k_0(\theta_0 - \tau_0) \quad [25]$$

$$(1 - m^2)E_1(\phi - \alpha_1\theta_1) = (1 - M^2)E_0(\phi - \alpha_0\theta_0) \quad [26]$$

$$\lambda_0[2E_1(\phi - \alpha_1\theta_1) + 2\sigma_\rho] = \lambda_1 2E_0(\phi - \alpha_0\theta_0) + 2\sigma_\rho] \quad [27]$$

a system of three equations for the three unknowns ϕ, θ_1 and θ_0.

To solve the system, we introduce a set of parameters δ_1, δ_0, μ_1 and μ_0, defined by

$$\theta_1 = \tau_1 + E_1\delta_1 \qquad \theta_0 = \tau_0 + E_0\delta_0 \quad [28]$$

$$\phi = \alpha_1\theta_1 - k_1\delta_1\mu_1 = \alpha_0\theta_0 - k_0\delta_0\mu_0 \quad [29]$$

Then Equation [25] becomes

$$(1 - m^2)E_1k_1\delta_1 = (1 - M^2)E_0k_0\delta_0 (= 2\sigma_\rho) \quad [30]$$

while Equation [26] gives

$$(1 - m^2)E_1k_1\delta_1\mu_1 = (1 - M^2)E_0k_0\delta_0\mu_0$$

whence, by comparison, $\mu_1 = \mu_0 = \mu$, say.

The substitution into Equation [27] of the appropriate expressions for ϕ and $2\sigma_\rho$ from Equations [29, 30] then yields

$$\lambda_0E_1k_1\delta_1[-2\mu + (1 - m^2)] = \lambda_1E_0k_0\delta_0[-2\mu + (1 - M^2)] \quad [31]$$

Put

$$2\mu = (1 - m^2) - (M^2 - m^2)\frac{\epsilon}{1 + \epsilon} = (1 - M^2) + (M^2 - m^2)\frac{1}{1 + \epsilon} \quad [32]$$

in terms of an auxiliary parameter ϵ. Then, from Equations [31, 30] we find that

$$\epsilon = \frac{\lambda_1(1 - m^2)}{\lambda_0(M^2 - 1)} \quad [33]$$

and hence from Equation [30] that

$$\delta_0 = -\epsilon\frac{\lambda_0E_1k_1}{\lambda_1E_0k_0}\delta_1 \quad [34]$$

Finally, δ_1 may be found in a straightforward way from Equations [28, 29, 34] in terms of the known parameters ϵ and μ (see below).

Collecting the results, we have

$$\left.\begin{aligned} d_1 &= \frac{1}{2}a^2k_1\theta_1 & d_0 &= \frac{1}{2}c^2k_0\theta_0 \\ \theta_1 &= \tau_1 + E_1\delta_1 & \theta_0 &= \tau_0 + E_0\delta_0 \\ \phi &= \alpha_1\theta_1 - k_1\delta_1\mu & &= \alpha_0\theta_0 - k_0\delta_0\mu \end{aligned}\right\} \quad [35]$$

where $2\mu = (1 - m^2) - (M^2 - m^2)\dfrac{\epsilon}{1 + \epsilon}$,

$$\epsilon = \frac{\lambda_1(1 - m^2)}{\lambda_0(M^2 - 1)}$$

and

$$\delta_1 = \lambda_1E_0k_0\left[\frac{\alpha_0\tau_0 - \alpha_1\tau_1}{\lambda_1E_0k_0(\alpha_1E_1 - k_1\mu) + \lambda_0E_1k_1(\alpha_0E_0 - k_0\mu)}\right]$$

$$\delta_0 = -\epsilon\frac{\lambda_0E_1k_1}{\lambda_1E_0k_0}\delta_1$$

The stress and displacement values can be evaluated in terms of θ_1, θ_0 and ϕ from Equation [35] if the temperature distribution is known. Since none of the parameters in Equation [35] is singular when $m = 0$, the stresses in a solid grain are included in the solutions found here, and are obtained by formally setting $m = 0$ in Equations [35, 24].[4]

Solutions for a Special Case

It is ordinarily true that the shell material is much "stiffer" than the plastic used for the grain, i.e., $E_0 \gg E_1$. Hence, if the shell is not too thin, $M = c/b$ will be sufficiently greater than unity for the parameter

$$\epsilon = \frac{\lambda_1(1 - m^2)}{\lambda_0(M^2 - 1)} = \frac{E_1}{E_0}\frac{(1 - m^2)}{(M^2 - 1)}\frac{(1 + \sigma_0)}{(1 + \sigma_1)} \text{ to be} \ll 1 \quad [36]$$

The last of Equation [35] then shows that δ_0/δ_1 is of order ϵ^2, and hence that the products $E_0\delta_0$, $k_0\delta_0$ are of order ϵ in comparison with $E_1\delta_1$, $k_1\delta_1$. The factors $\lambda_1E_0k_0(\alpha_1E_1 - k_1\mu)$, $\lambda_0E_1k_1(\alpha_0E_0 - k_0\mu)$ in the denominator of δ_1 (or δ_0) are about of the same order of magnitude, but the latter is multiplied by an additional factor of ϵ in Equation [35].

When ϵ is sufficiently small, one may to a first approximation neglect terms of relative order ϵ in Equation [35]. The equations then yield

$$\left.\begin{aligned} &\delta_0 = 0;\ \theta_0 = \tau_0;\ \phi = \alpha_0\theta_0 = \alpha_0\tau_0;\ \mu = \tfrac{1}{2}(1 - m^2) \\ &\delta_1 = \frac{\alpha_0\tau_0 - \alpha_1\tau_1}{\alpha_1E_1 - k_1\mu} = \frac{2}{k_1}\frac{\alpha_0\tau_0 - \alpha_1\tau_1}{2(1 - \sigma_1) - (1 - m^2)} \\ &\alpha_1\theta_1 = \alpha_1\tau_1 + \alpha_1E_1\delta_1 = \frac{2(1 - \sigma_1)\alpha_0\tau_0 - (1 - m^2)\alpha_1\tau_1}{2(1 - \sigma_1) - (1 - m^2)} \end{aligned}\right\} \quad [37]$$

Using the definition of ϕ and θ_1 in Equation [37] (and $\alpha_1E_1 = (1 - \sigma_1)k_1$), one obtains for the combination appearing in σ_z

$$E_1(\phi - \alpha_1\theta_1) + k_1\theta_1 = E_1\alpha_0\tau_0 + (k_1 - \alpha_1E_1)\theta_1 = \frac{\alpha_0}{\alpha_1}k_1(1 - \sigma_1)\tau_0 + k_1\sigma_1\theta_1 =$$

$$k_1\sigma_1\theta_1 + \frac{k_1}{2\alpha_1}[(1 - m^2)\alpha_1\tau_1 + (1 - 2\sigma_1 + m^2)\alpha_1\theta_1] = \frac{1}{2}k_1[(1 - m^2)\tau_1 + (1 + m^2)\theta_1]$$

In the approximation considered, the formulas for the stresses in the grain thus reduce to

$$\left.\begin{aligned} \sigma_\rho &= \frac{1}{2}k_1\tau_0\left[\left(1 - \frac{a^2}{\rho^2}\right)\left(\frac{\theta_1}{\tau_0}\right) - \left(\frac{\bar{T}}{\tau_0}\right)\right] \\ \sigma_t &= \frac{1}{2}k_1\tau_0\left[\left(1 + \frac{a^2}{\rho^2}\right)\left(\frac{\theta_1}{\tau_0}\right) + \left(\frac{\bar{T}}{\tau_0}\right) - 2\left(\frac{T}{\tau_0}\right)\right] \\ \sigma_z &= \frac{1}{2}k_1\tau_0\left[\left(1 + \frac{a^2}{b^2}\right)\left(\frac{\theta_1}{\tau_0}\right) + \left(1 - \frac{a^2}{b^2}\right)\left(\frac{\tau_1}{\tau_0}\right) - 2\left(\frac{T}{\tau_0}\right)\right] \end{aligned}\right\} \quad [38]$$

Here, τ_1 and τ_0 give the average temperature in the grain and

[4] The stresses in a free (i.e., not case-bonded) grain may similarly be obtained by letting M tend to 1. This case has been discussed by Geckler (Reference 2), on the basis of temperature solutions presented by Jaeger (Reference 3).

shell, respectively, T is the actual temperature at the point for which Equation [38] is written

$$\left.\begin{aligned} \bar{T} = \bar{T}(\rho) &= \frac{2}{\rho^2}\int_a^\rho xT(x)dx \\ k_1 &= \frac{\alpha_1 E_1}{1-\sigma_1} \end{aligned}\right\} \quad \text{.....[39]}$$

The case of a solid grain is obtained by formally setting $a = m = 0$ in these equations.

Initial and Final Stresses From [38]

The formulas developed above give the stress distribution, at any instant of time, in terms of the temperature distribution at that instant. What one usually requires, however, is not the instantaneous stress distribution but the maximum stresses—in order to determine the possibility of failure of the case-bonding, for instance, or the possibility of cracks developing in the grain. For the problem considered in the introduction, it appears that the maximum stresses in the grain will occur either initially or finally.[5]

[5] By "initially" here, we mean at time $t = 0^+$, say—after the (metallic) shell has come to thermal equilibrium at temperature T_0, but before significant heat transfer has taken place in the grain. By "finally" we mean after the grain has come to thermal equilibrium, but before appreciable stress relaxation due to viscous flow has occurred in the grain (see the Introduction).

The heat problem then reduces to that of a concentric annular cylinder, thermally insulated at its inner surface, which is in thermal equilibrium at time $t = 0$. At $t = 0^+$, the temperature at its outer surface is suddenly increased by an amount T_0. Thus

$$\dot{T} \equiv \frac{1}{D}\frac{\partial T}{\partial t} = \frac{\partial^2 T}{\partial \rho^2} + \frac{1}{\rho}\frac{\partial T}{\partial \rho} \quad (t > 0^+,\ a \le \rho < b)$$

$$\frac{\partial T}{\partial \rho} = 0 \text{ at } \rho = a,\ T = T_0 \text{ at } \rho = b;\ T = 0 \text{ initially } (a \le \rho < b)$$

where D represents the thermal diffusivity of the grain material and we have for brevity denoted $(1/D)(\partial/\partial T)$ by a dot.

For the statement above to be true, the stresses must change monotonically with time. It is sufficient to consider the case $T_0 > 0$.

Since $\dot{T}_0 = 0$ and $\dot{T} \ge 0$ for $t > 0^+$, it is easily shown that the quantities

$$\dot{\bar{T}} = \frac{2}{\rho_2}\int_a^\rho x\,\dot{T}\,dx = \frac{2}{\rho}\frac{\partial T}{\partial \rho},\ (1-m^2)\dot{r}_1 = \dot{T}(b),\ -\dot{\theta}$$

and

$$-(1+m^2)\dot{\theta} - (1-m^2)\dot{r}_1$$

will be increasing functions of time, and hence that after $t = 0^+$ (after the initial stresses due to shell expansion are applied), the longitudinal and radial stresses will decrease: $\dot{\sigma}_\rho,\ \dot{\sigma}_z \le 0$. It is a more subtle matter to show that the tangential stress is also a decreasing function of time (unless $m^2 > 1 - 2\sigma_1$). However, it appears that, for the problem considered, $\partial^2 T/\partial \rho^2$ is either positive, or else has its least value at $\rho = b$. In either case

$$2\frac{\partial^2 T}{\partial \rho^2} = 2\dot{T} - \dot{\bar{T}} \ge [2\dot{T} - \dot{\bar{T}}]_{\rho=b} = -\dot{\bar{T}}(b) = -(1-m^2)\dot{r}_1$$

Then

$$\dot{\sigma}_t = \frac{1}{2}k_1\left[\left(1+\frac{\rho^2}{a^2}\right)\dot{\theta}_1 + \dot{\bar{T}} - 2\dot{T}\right] \le \frac{1}{2}k_1\left[\left(1+\frac{a^2}{\rho^2}\right)\theta_1 + (1-m^2)\dot{r}_1\right]$$

$$= \frac{1}{2}k_1(1-m^2)\dot{r}_1\left[1 - \frac{1+\frac{a^2}{\rho^2}}{1-2\sigma_1+m^2}\right]$$

$$= \frac{1}{2}k_1(1-m^2)\dot{r}_1\left[\frac{m^2 - 2\sigma_1 - \frac{a^2}{\rho^2}}{1-2\sigma_1+m^2}\right]$$

which is negative for $0 < \sigma_1 < \frac{1}{2}$, $0 < a \le \rho \le b$.

At time $t = 0^+$, $T = 0$ for $\rho < b$, while $T = T_0\ (=\tau_0)$ at $\rho = b$. Thus, $\bar{T} = \tau_1 = 0$, and

$$\alpha_1\theta_1 = \frac{2(1-\sigma_1)\alpha_0 T_0}{2(1-\sigma_1) - (1-m^2)} \quad \text{.....[40a]}$$

giving for the initial stresses

$$\left.\begin{aligned} \sigma_\rho &= \frac{1}{2}k_1T_0\left(\frac{\theta_1}{T_0}\right)\left(1-\frac{a^2}{\rho^2}\right) \\ \sigma_t &= \frac{1}{2}k_1T_0\left(\frac{\theta_1}{T_0}\right)\left(1+\frac{a^2}{\rho^2}\right) \\ \sigma_z &= \frac{1}{2}k_1T_0\left(\frac{\theta_1}{T_0}\right)(1+m^2) \end{aligned}\right\} \quad \text{.....[40b]}$$

for $\rho < b$ $(t = 0^+)$, while

$$\left.\begin{aligned} \sigma_\rho &= \frac{1}{2}k_1T_0\left(\frac{\theta_1}{T_0}\right)(1-m^2) \\ \sigma_t &= \frac{1}{2}k_1T_0\left[\left(\frac{\theta_1}{T_0}\right)(1+m^2) - 2\right] \\ \sigma_z &= \frac{1}{2}k_1T_0\left[\left(\frac{\theta_1}{T_0}\right)(1+m^2) - 2\right] \end{aligned}\right\} \quad \text{.....[40c]}$$

at $\rho = b$ $(t = 0^+)$.

At time $t = \infty$, $T = \tau_1 = T_0\ (=\tau_0)$, so that

$$\alpha_1\theta_1 - \alpha_1\tau_0 = \frac{2(1-\sigma_1)(\alpha_0 - \alpha_1)T_0}{2(1-\sigma_1) - (1-m^2)} \quad \text{.....[41a]}$$

The final stresses therefore reduce to

$$\left.\begin{aligned} \sigma_\rho &= \frac{1}{2}k_1T_0\left[\frac{\theta_1}{T_0} - 1\right]\left(1-\frac{a^2}{\rho^2}\right) \\ \sigma_t &= \frac{1}{2}k_1T_0\left[\frac{\theta_1}{T_0} - 1\right]\left(1+\frac{a^2}{\rho^2}\right) \\ \sigma_z &= \frac{1}{2}k_1T_0\left[\frac{\theta_1}{T_0} - 1\right](1+m^2) \end{aligned}\right\} \quad \text{.....[41b]}$$

for $a \le \rho \le b$ $(t \to \infty)$. If the grain and shell should happen to have the same linear coefficient of thermal expansion, the stresses would evidently have to vanish when the system reaches thermal equilibrium. That this will be the case (for the model used) may be seen from Equations [41a, 41b]. When $\alpha_0 = \alpha_1$, Equation [41a] gives $\theta_1 = \tau_0 = T_0$, and therefore the stresses in Equation [41b] vanish identically.

Acknowledgment

The author wishes to thank Richard D. Geckler of Aerojet-General Corp. for suggesting the problem discussed above, and for his encouragement and constructive criticism during the course of the investigation.

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Investigation of the Factors Affecting the Attachment of a Liquid Film to a Solid Surface[1]

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One method of establishing a protective liquid film between a flowing gas and the walls of the flow passage is by the injection of the liquid through a slot in the passage walls. The effects of liquid injection velocity, gas velocity, slot geometry, liquid physical properties and gas density upon the film attachment are examined experimentally. A correlating equation based upon momentum exchange for the case of radial injection is presented.

Nomenclature

b = slot width
D = test section diameter or 4 times hydraulic radius
V_L = velocity of liquid through slot
V_L^* = critical velocity of injection
V_a = average air velocity in test section
Re_g = air Reynolds number
α = angle of injection slot
ρ_L = liquid density
ρ_g = gas density
σ = liquid surface tension
μ = liquid or gas viscosity

Introduction

THE phenomena of the flow of a thin layer of liquid, hereafter termed "film flow," over a solid service is encountered in many engineering processes. In the rocket propulsion field thin layers of liquid have been used as "a film coolant" to protect the inner surfaces of rocket motors from extremely hot combustion gases (1–3).[4] The external surfaces of high velocity missiles might also be protected from "aerodynamic heating" by the use of a thin layer of liquid as a film coolant. As part of a fundamental investigation of the phenomenon of film flow at the Jet Propulsion Center of Purdue University, an investigation, initiated in 1948, has been conducted of the parameters which affect the attachment of a liquid film to the inside surface of ducts through which air is flowing (4–9). In the experimental program at Purdue the liquid film has been produced by injecting the liquid through spaced parallel disks, hereafter termed the "slot." The effect of various liquid and gas physical properties and various physical dimensions on the "critical velocity of injection"[5] has been determined. The purpose of this paper is to summarize the results of the experimental investigation on the film attachment phase of the film flow study and to present an empirical equation based on momentum exchange that correlates the results of the investigation with two- and three-dimensional flow fields.

Investigation of film attachment was carried out in both two- and three-dimensional flow fields. In the two-dimensional test section the liquid was injected through a slot in the bottom of the test section. In the three-dimensional test section the liquid was injected through a slot that was formed on the inner periphery of the duct.

The parameter selected as the criteria of film attachment was the critical velocity of injection. At a constant air velocity the critical velocity of injection V_L^* was determined by gradually increasing the velocity of injection until the liquid film was observed to separate from the downstream side of the slot. At liquid velocities less than the critical velocity of injection the injected film is caused to bend and flow along the solid surface by the physical action of the gas stream without separating from the solid surface. At some velocity of

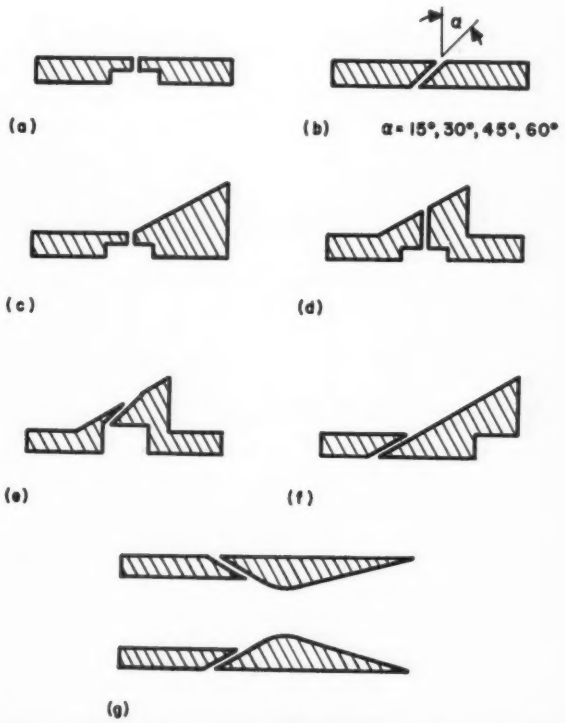

Fig. 1 Slot configurations

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[3] Associate Professor of Mechanical Engineering. Mem. ARS.

[4] Numbers in parentheses indicate References at end of paper.

[5] The "critical velocity of injection" is defined as the mean liquid velocity flowing through the injection slot corresponding to the maximum rate of liquid flow obtained with no visible separation of the liquid film from the surface of the test section, hereafter referred to as V_L^*.

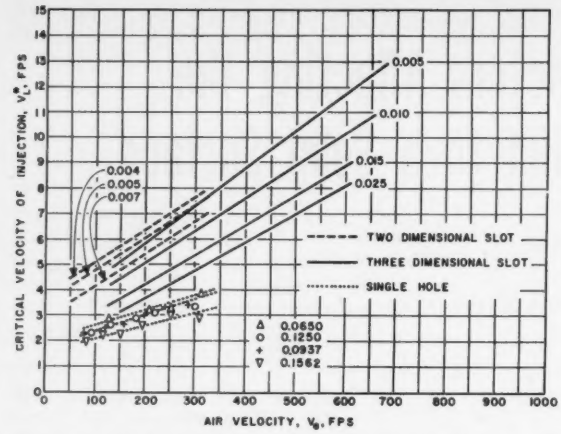

Fig. 2 Effect of air stream velocity and slot width upon the critical velocity of injection

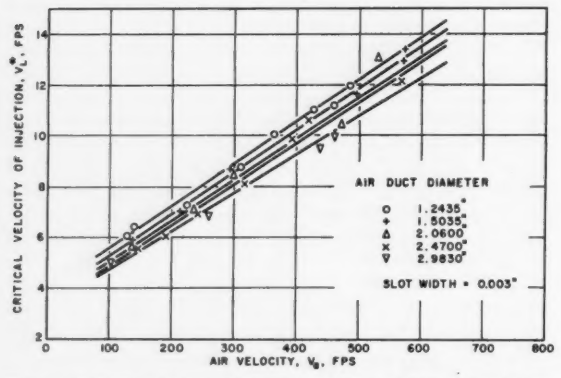

Fig. 3 Critical velocity of injection vs. air velocity showing the effect of duct diameter

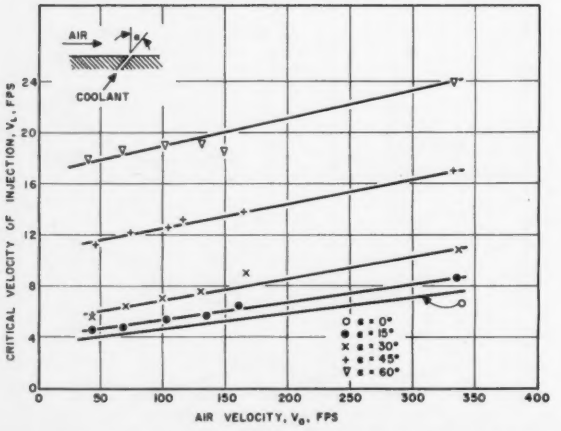

Fig. 4 Effect of air velocity upon critical velocity of injection for various angles of injection

injection the liquid film separates momentarily from the surface, but is reattached just downstream from the slot. The liquid velocity at which the separation occurs is termed the critical velocity of injection. At still higher liquid velocities the film separates completely from the solid surface and mixes with the gas stream.

The determination of V_L^* was made visually and consequently it was necessary to develop experimental techniques to keep the human error as small as possible. Check runs on reproducibility were made in all of the experiments and the reproducibility of the results from day to day for a single observer, and between different observers, was of the order of ± 10 per cent. The effects of hysteresis and gravity were at no time large enough to cause significant differences or warrant special attention.

In the two-dimensional investigation a number of the physical dimensions were varied: (a) Test section height, 3 and 5 in.; (b) slot width, 0.001, 0.003, 0.004, 0.005 and 0.007 in.; (c) slot depth, $\frac{1}{8}$ and $\frac{1}{4}$ in.; and (d) injection configurations as shown in Fig. 1. In all of the above experiments the test section width was $1\frac{1}{2}$ in. and the slot length was 1 in.

In the three-dimensional investigation (a) the duct diameters were 1.2435, 1.503, 2.0600, 2.470 and 2.9830 in.; (b) the slot widths were 0.005, 0.010, 0.015 and 0.025 in.; and the configurations listed as (a, c, d, e, f, g) in Fig. 1 were tested. In both the two- and three-dimensional investigation the velocity profile of the air at the test section was fully developed.

Effect of Physical Dimensions on Critical Velocity of Injection

Fig. 2 presents V_L^* as a function of V_a for several slot widths and hole diameters using water as the injected liquid. It is seen that for any slot width or hole diameter increasing the air velocity increases V_L^* and the relationship between V_L^* and V_a is approximately linear. Investigations with carbon tetrachloride and 40 per cent aqueous glycerol solutions and different physical configurations substantiate these results.

Fig. 2 also illustrates the effect of slot width in the two- and three-dimensional test sections for the case of water injection at right angles to the air stream. The results of the investigation using holes conducted by Knuth (11) is also included for comparison purposes. No comparison can be made between slots and holes, but it should be noted that the trend is the same; that is, the larger the hole or slot, the lower V_L^* is for a given air velocity.

Preliminary investigations conducted in the two-dimensional test section of variation in test section dimensions indicated that decreasing the height of the section from 5 to 3 in. increased V_L^* approximately 10 per cent. A systematic investigation on the effect of test section diameter was conducted in the three-dimensional test section. Fig. 3 presents the critical velocity of injection as a function of air velocity for five duct diameters all having a slot width of 0.005 in. The critical velocity of injection increased as the duct diameter decreased. Similar results were obtained for the five duct diameters with slot widths of 0.010, 0.015 and 0.025. in.

While most of the studies of film attachment were conducted with perpendicular injection (radial injection), an investigation of the effect of the angle of injection, denoted by α, was conducted in the two-dimensional test section with the injector slots shown in Fig. 1 (b). Some results on the effect of angle of injection were also indicated by the experiments with the two- and three-dimensional configurations, (d), (e) and (f) of Fig. 1. Fig. 4 presents the critical velocity of injection as a function of air velocity for values of $\alpha = 0$, 15, 30 and 60 deg. Perpendicular injection corresponds to a value

of $\alpha = 0$ deg. As would be expected, inclining the film flow in the downstream direction increases the critical velocity of injection. At injection angles corresponding to values of α greater than 75 deg it was not possible with the experimental apparatus to separate the film from the surface even when there was no flow of air. One explanation that has been suggested for firm attachment of the film at angles of α greater than 75 deg is the Coanda effect (14).

No method of correlating the effect of angle based on the geometry of the system was possible. However, if the "effective" angle was used as $\alpha + 15$ deg (suggested by the fact that if injection was made at angles of 75 deg or larger the injector acted as if it were injected parallel to the surface), it was possible to predict the values of critical velocity of injection for the two-dimensional experiments at angles of α from 15 to 60 deg by dividing the critical velocity of injection for the perpendicular case by $\cos(\alpha + 15 \text{ deg})$.

Fig. 5 presents the experimental and "predicted" values of V_L^* as a function of angle for three air velocities. The agreement is satisfactory for these cases and for all two-dimensional results. However, for the three-dimensional investigation the above method does not predict the experimental values.

Nozzle configurations (c), (d), (e) and (f) presented in Fig. 1 were tested in both the two- and three-dimensional test sections. Configuration (c), which simulates radial injection at the junction of the combustion chamber and nozzle, had a V_L^* slightly smaller than configuration (a) in the two-dimensional test section, but had a higher V_L^* in the three-dimensional test section. The improved performance in the three-dimensional test section is attributed to the uniform construction of the flow field which should cause a more uniform flow pattern at the junction than in the two-dimensional case where the flow field was only restricted on one side.

The critical velocity of injection of configuration (d) in both the two- and three-dimensional case was higher than that of (a) or (c). The higher critical velocities of injection may be attributed to the higher value of α (30 deg) as compared to configuration (a). Again, as in the case of configurations (b), it is possible to estimate the performance of configurations (d) for the two-dimensional case by dividing the zero angle injection value by cos (30 + 15) deg. Configuration (d) could be operated at slightly supercritical liquid injection velocities to form stable liquid films a short distance downstream from the injection slot.

Configurations (e) and (f) for both the two- and three-dimensional cases formed extremely stable films. It was not possible to produce flow separation at a V_L^* as high as 60 fps, the maximum obtainable with the experimental apparatus.

The investigation of the three-dimensional flow of liquid films in a 60 deg converging-30 deg diverging passage was conducted with subsonic air velocities throughout. The film remained on the walls through the converging throat and a portion of the diverging section. The film separated at the point where the air separated from the diverging section. Under conditions where the air did not separate the film adhered throughout the passage.

Effect of Physical Properties of the Liquid on Critical Velocity of Injection

Aqueous soap solutions, aqueous glycerol solutions and aqueous zinc chloride solutions were employed as injection liquids in the two-dimensional test apparatus in a study of the effects of liquid physical properties upon V_L^*. The radial slot width was maintained constant at 0.005 in. during all experiments. The pertinent physical properties of the four liquids employed are tabulated in Table 1.

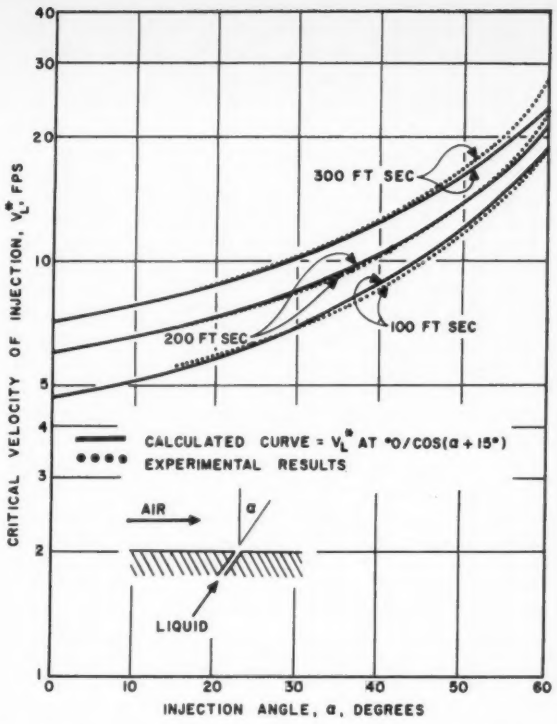

Fig. 5 Effect of angle of injection upon critical velocity of injection for three air velocities

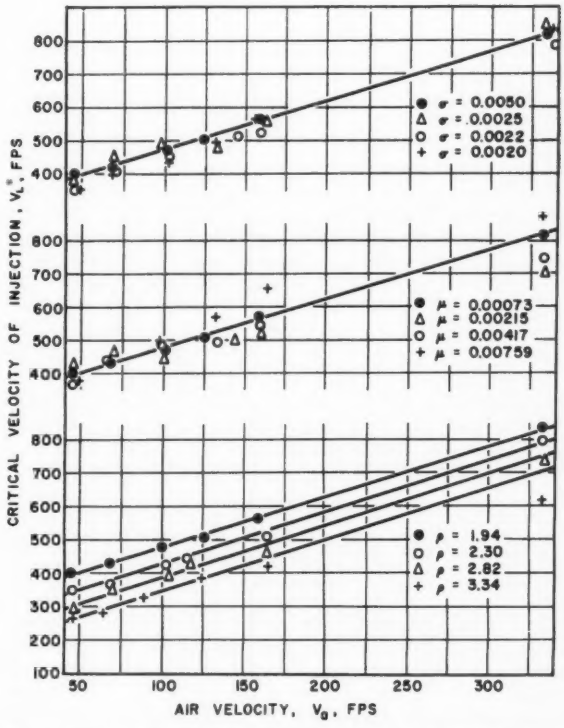

Fig. 6 Effect of liquid properties upon critical velocity of injection for various air stream velocities

Table 1 Selected physical properties of test liquids

(Composition by weight)

Liquid	Density, slugs/ft³	Viscosity, lb/ft sec	Surface tension, lb/ft
Water	1.94	0.000734	0.005
Soap solution	1.94	0.000827	0.0025
Soap solution	1.94	0.000793	0.0022
Soap solution	1.94	0.000793	0.0020
32% glycerol	2.10	0.00215	
44% glycerol	2.17	0.00417	
58% glycerol	2.23	0.00759	
22.6% zinc chloride	2.30	0.00122	
45.4% zinc chloride	2.82	0.0023	
58.5% zinc chloride	3.34	0.00785	

The effect of changes in liquid surface tension upon V_L^* was examined by plotting V_L^* as a function of main air stream velocity V_a, employing the data obtained from experiments conducted with water and the aqueous soap solutions as shown in Fig. 6. Since the densities and viscosities of the three soap solutions and of water are of the same order of magnitude, it was believed any consistent variation in results could be attributed to the change in liquid surface tension. From Fig. 6 it is seen that no consistent variation was obtained; therefore, it may be concluded that variations in surface tension of the injected liquid have no appreciable effect upon V_L^*.

There is some evidence to indicate that the results obtained by employing soap to change the liquid surface tension may not be the same as those obtained using pure liquids having different surface tensions (12).

Fig. 6 also presents a plot of V_L^* as a function of V_a obtained when employing water and the aqueous glycerol solutions as the injected liquid. Since the densities and the surface tensions of those liquids were considered constant, any consistent variation in the results may be attributed to change in liquid viscosity. As can be seen from Fig. 6, the straight line representing V_L^* as a function of V_a for water represents the approximate average of all of the data points obtained with the glycerol solutions; therefore, it was concluded that liquid viscosity has little or no effect upon V_L^*. It should not be overlooked, however, that the data points obtained with the different glycerol solutions are displaced slightly from those obtained with water.

Unfortunately, changes in test liquid density were accompanied by variations in liquid viscosity and surface tension. Since it had been demonstrated previously that changes in both liquid viscosity and surface tension have negligible influence upon V_L^*, it was believed that any variation in the data obtained employing the zinc chloride solution could be attributed to changes in liquid density. Referring to Fig. 6 it is seen that the plot of the test results gave four distinct parallel lines exhibiting a uniform change in V_L^* with changes in liquid density, thus indicating a significant dependence of V_L^* upon liquid density.

Effect of Gas Density on Critical Velocity of Injection

An experimental study was made of the effect of gas density upon V_L^* employing the three-dimensional test apparatus having an inside duct diameter of 2.02 in. with a 0.005-in.-wide radial slot. Employing water as the test liquid, variations in air density from approximately 0.0023 slugs/ft³ to 0.0795 slugs/ft³ were obtained by altering the static pressure of the air flowing in the test section. Experiments were performed for mean air velocities ranging from 48 to 617 fps resulting in a range of air stream Reynolds numbers from 1.67×10^5 to 14.75×10^5.

The experimental results indicate that, in general, V_L^* increased markedly with increasing air density.

An attempt was made to study the effect of air stream viscosity upon V_L^* by employing the heated combustion products of air and propane as the flowing gas. Unfortunately, because of the manner in which the experiments were performed, a variation in gas density as well as a variation in gas viscosity occurred and no definite conclusions concerning the dependence of V_L^* upon gas stream viscosity could be reached. It is believed, however, that the effect of gas viscosity is minor.

Correlation Studies

In order to organize the results of the extensive experimental program discussed herein in a manner that would be of benefit to the designer of equipment utilizing liquid films, numerous attempts were made to obtain suitable dimensionless correlations of the experimental data. No satisfactory correlation was obtained which included all of the geometric configurations for the liquid injection studies; therefore, only the data from the radial injection experiments were utilized.

In obtaining the correlating equation it was assumed that the bending of the liquid film away from the radial direction of injection was due primarily to the momentum interchange between the air stream and the injected liquid. The distance of liquid film penetration into the air stream was assumed to be a function of both the air stream Reynolds number and slot width. Combining the momentum parameter with the slot width parameter b/D and the air stream Reynolds number gives the functional relationship

$$f\left(\frac{V_L^*}{V_a}\sqrt{\frac{\rho_L}{\rho_a}}, \frac{b}{D}, Re_a\right) = 0 \quad \ldots\ldots\ldots\ldots [1]$$

The resulting correlation is presented in Fig. 7. It can be seen from Fig. 7 that two equations, each representing a straight line on a log plot, are required to correlate the data; the straight lines intersect at a Reynolds number of 2.5×10^5. The two equations discussed above are

$$\frac{V_L^*}{V_a}\sqrt{\frac{\rho_L}{\rho_a}}\left(\frac{b}{D}\right)^{f'(Re_a)} = 0.153\,[Re_a \times 10^{-5}]^{-0.65} \quad \ldots [2]$$

for gas Reynolds numbers from 0.5×10^5 to 2.5×10^5, and

$$\frac{V_L^*}{V_a}\sqrt{\frac{\rho_L}{\rho_a}}\left(\frac{b}{D}\right)^{f'(Re_a)} = 0.085 \quad \ldots\ldots\ldots\ldots [3]$$

for gas Reynolds numbers from 2.5×10^5 to 12×10^5 where $f'(Re_a) = 0.404 - 0.012Re_a(10)^{-5}$.

It was observed that the height of penetration of the injected liquid film is many times greater for large slot widths and low air Reynolds numbers than for smaller slot widths and Reynolds numbers. For the above reason it is believed that the injection film is more likely to enter the turbulent region of the air stream boundary layer under conditions of large slot width and low air Reynolds number, and it has been found that these conditions are most prevalent in the portion of Fig. 7 represented by Equation [2].

The data obtained from the experiments performed over a range of air densities indicate that the fluid density term $(\rho_L/\rho_a)^{1/2}$ in Equations [2, 3] should be replaced by $(\rho_L/\rho_a)^{0.5}$ to obtain a better correlation.

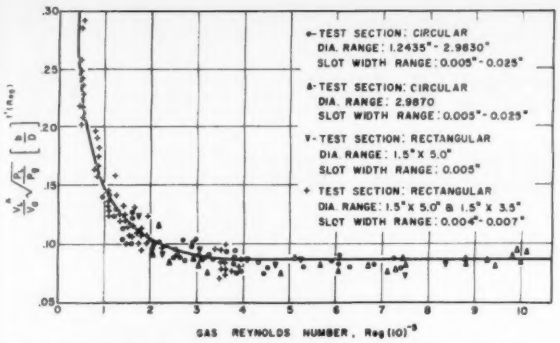
TEST SECTION: CIRCULAR
DIA. RANGE: 1.2435" - 2.9830"
SLOT WIDTH RANGE: 0.005" - 0.025"
TEST SECTION: CIRCULAR
DIA. RANGE: 2.9870
SLOT WIDTH RANGE: 0.005" - 0.025"
TEST SECTION: RECTANGULAR
DIA. RANGE: 1.5" X 5.0"
SLOT WIDTH RANGE: 0.005"
TEST SECTION: RECTANGULAR
DIA. RANGE: 1.5" X 5.0" & 1.5" X 3.5"
SLOT WIDTH RANGE: 0.004" - 0.007"
GAS REYNOLDS NUMBER, Re_g (10)^-3

Fig. 7 Dimensionless correlations of critical velocity of injection data

Visual Studies of Liquid Film

Visual and photographic studies of the injection slot revealed the presence of disturbances on the liquid film surface which moved at velocities greatly different from the main film velocity. Such disturbances have also been reported by Kinney, Abramson and Sloop (13), and by Knuth (11). The velocity of the liquid film free surface and the velocity of the disturbances moving on the film surface were measured by high-speed photographic techniques employing a dye injected into the water film. The velocities of the disturbances were found to be considerably larger than the free surface velocity of the liquid film; at low velocities of injection the velocities differed by as much as a factor of 3. The wavelike nature of the surface disturbances was found to be independent of the manner of liquid injection. The photographic study further indicated that the magnitude and number of the surface disturbance increased when (a) the liquid flow rate was increased and (b) the air velocity was decreased.

A phenomenon which may have practical significance for film-cooling applications was observed when the radial injection slots were operated at high air velocities. When liquid injection velocity exceeded 90 per cent of V_L^*, a portion of the liquid admixed with the air stream, although no film separation occurred.

In the case of liquid film coolant applications, it should be pointed out that in view of the above described phenomena and the formation of wave disturbances upon the liquid film surface, both occurring at injection velocities near the critical velocity of injection, the liquid injection velocity should not be more than 90 per cent of the critical velocity of injection for the most efficient utilization of the available liquid.

Conclusions

The pertinent conclusions summarizing the results of the series of experiments presented in the paper are:

1 The injection of liquid through slots results in the establishment of stable films attached to solid surfaces.

2 The critical velocity of injection V_L^*:

a. Increases with increases in air velocity V_a.

b. Decreases with increases in slot width b.

c. Increases with increases in angle of injection α; separation could not be obtained with angles larger than 75 deg.

3 The value of V_L^* is independent of liquid surface tension and viscosity but is a function of liquid density (momentum).

4 The value of V_L^* is a function of air density (momentum).

5 The use of large velocities of injection, below but approaching V_L^*, results in the formation of surface disturbances upon the film surface, and the entrainment of a considerable portion of the injected liquid by the air stream.

Acknowledgments

The authors wish to express their appreciation to C. M. Beighley, E. L. Knuth, A. B. Greenberg, J. P. M. Diamond, R. K. Louden and D. A. Charvonia, graduate students at Purdue University, who conducted the experimental research summarized in this paper. Appreciation is also expressed to M. J. Zucrow who directed the research and made many helpful suggestions in the preparation of the paper.

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Temperatures and Thermal Stresses in Transpiration-Cooled Power-Producing Plates and Tubes

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Solutions are given for the local temperature, mean temperature, efficiency, surface temperature difference and thermal stress in transpiration-cooled porous plates and tubes generating uniform distributed heat. Eight special cases are considered, four of which refer to the conventional case of nongenerating porous materials.

Nomenclature

A = $(1/M - 1)N_g/(N_g + 1)$, dimensionless
a = $G_cC_c\delta/(1 - p)k_w$, dimensionless
B = $A - 1/a$, dimensionless
b = $G_{cg}C_cr_g/(1 - p)k_w$, dimensionless
C_c = thermal capacity of coolant, Btu/lb-°F
C_1, C_2 = integration constants
E = modulus of elasticity, lb/ft²
G_c = mass velocity of coolant, lb/hr-ft²
G_{cg} = mass velocity of coolant at exit side, lb/hr-ft²
h = unit surface conductance, Btu/hr-ft²-°F
k_w = thermal conductivity of porous wall material, Btu/hr-ft-°F
M = $(\lambda/a)/(t_g - t_c)$ or $(\lambda/a)/(t_o - t_c)$, dimensionless
N_c = h_c/G_cC_c for plate, $h_c/G_{cg}C_c$ for tube, dimensionless
N_g = h_g/G_cC_c for plate, $h_g/G_{cg}C_c$ for tube, dimensionless
P = $(\mu/b)/(t_g - t_c)$ or $(\mu/b)/(t_o - t_c)$, dimensionless
p = porosity, dimensionless
q_w''' = internal heat-generation rate, Btu/hr-ft³
R = r/r_g, dimensionless
R_c = r_c/r_g, dimensionless
r = local tube radius, ft
r_c = radius of outer tube surface, ft
r_g = radius of inner tube surface, ft
t = local temperature, °F
$\bar{t}$ = mean temperature, °F
X = x/δ, dimensionless
x = local plate coordinate, ft
β = coefficient of linear thermal expansion, ft/ft-°F
δ = plate thickness, ft
η_1 = $(t_g - t_o)/(t_g - t_c)$, dimensionless
η_2 = $(t_o - t_b)/(t_g - t_c)$ for plate, $(t_o - t_1)/(t_g - t_c)$ for tube, dimensionless
λ = $q_w'''\delta^2/k_w$, °F
μ = $q_w'''r_g^2/k_w$, °F
ν = Poisson's ratio, dimensionless
σ = stress, lb/ft²
σ^* = $(1 - \nu)\sigma/E\beta$, °F

Subscripts

c = coolant
g = gas
o = exit surface of plate or tube
r = radial
t = tangential
w = wall material
y = lateral
z = longitudinal
δ = entrance surface of plate
1 = entrance surface of tube

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Introduction

THE analytical problem of predicting the effectiveness of transpiration-cooled structures consists of two parts, the first being concerned with the heat transfer and temperature distribution within the porous material, and the second dealing with the evaluation of convective unit surface conductances at the coolant-entry and coolant-exit side of the wall. In connection with the first problem, Weinbaum and Wheeler (1)[2] have given a solution for the solid and coolant temperatures within the porous matrix of a plane plate and a circular tube for the special case of negligible thermal resistance at the entrance and exit surfaces ($h_c = \infty = h_g$). These special solutions indicated that as a result of the large specific area for heat transfer within the wall—characteristic to the interstitial structure of the porous material—the solid and coolant temperatures become substantially equal to one another within only a few pore diameters from the coolant-entry surface.

Some consequences of this result have been discussed by Eckert (2). Mayer and Bartas (3) utilized this result as an explicit assumption in an over-all heat balance that allowed extension of the earlier solutions to include the effects of convective heat transfer at the two exposed surfaces. Although these new solutions were developed for both plane plates and circular tubes, the parameters involved did not disclose a dependence on the porosity of the wall material. In addition, due to an error in deriving the integration constants for the tube, the results for this geometry were incorrect. At about this same time a similar solution was presented (4) for a plane plate neglecting both the surface conductance at the coolant-entry side and the thermal resistance at the exit surface ($h_c = 0$, $h_g = \infty$). Recently Hyman (5) gave a solution for the plane plate, neglecting effects of porosity and of entrance surface conductance ($h_c = 0$). This represents a special case contained in the earlier plate solutions of Mayer and Bartas (3). Green (6) has given a solution for transpiration cooling of a plane plate generating uniform heat q_w''' within the solid material, neglecting coolant-side convection and assuming an adiabatic exit surface ($h_c = 0 = h_g$). The same solution including porosity effects is given in (4), and corresponding numerical solutions for this case are indicated in (7).

Whereas the chief purpose of the transpiration scheme is to provide effective cooling of structural elements exposed to high temperature gases, a secondary use of porous generating materials is for heating elements that combine high power density with compactness of heat exchange surface. Thus, the entrance fluid may be used either as a coolant to remove generated heat or as a gas to be heated by the high temperature porous material. In this way two admissible conditions arise: $t_c > t_g$ and $t_c < t_g$.

The purpose of this paper is to present general temperature and thermal stress solutions for porous generating structures assuming only that the coolant and solid temperatures are

[2] Numbers in parentheses indicate References at end of paper.

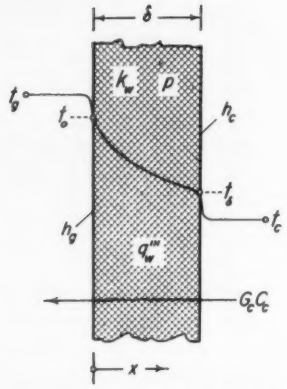

Fig. 1 Porous, heat-generating, transpiration-cooled plate

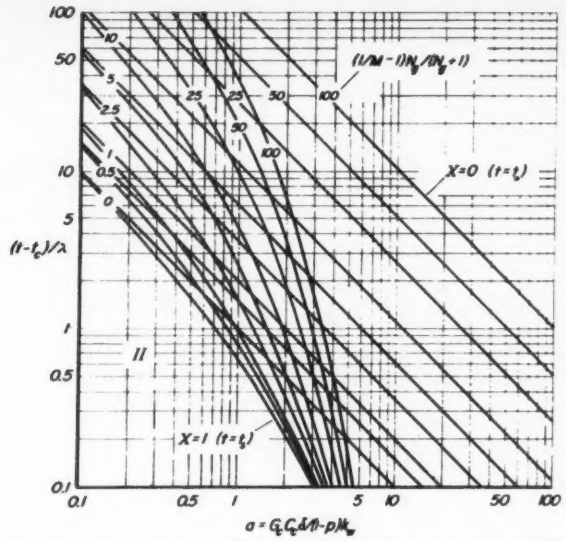

Fig. 2 Porous plate surface temperatures t_o and t_δ for Case II ($h_c = 0$)

equal at every point within the wall. Both plane plates and circular tubes are considered, and the solutions include the effects of heat generation within the solid, the porosity of the wall material, and the convective conductances at both exposed surfaces. The new solutions therefore contain as special cases the earlier results given in (3), (4), (5), and (6) along with the corrected tube results replacing those in (3). Formulas are given for (a) local wall temperature, (b) mean wall temperature, (c) efficiency, (d) surface temperature difference and (e) thermal stress. Some special cases for which explicit results are given are denoted as

I.	$h_c = \infty = h_g$	V.	$q_w''' = 0$
II.	$h_c = 0$	VI.	$q_w''' = 0, h_c = \infty = h_g$
III.	$h_c = 0, h_g = \infty$	VII.	$q_w''' = 0 = h_c$
IV.	$h_c = 0 = h_g$	VIII.	$q_w''' = 0 = h_c, h_g = \infty$

Analysis

Plane Plates

General Solution. Consider, as in Fig. 1, a plane plate of uniform thickness δ, uniform thermal conductivity k_w, and uniform porosity p (ratio of pore to total volume), generating uniform distributed heat q_w''' within the solid. A coolant of uniform thermal capacity C_c and uniform upstream temperature t_c is forced through the plate in the negative x-direction with a steady mass velocity G_c. The unit surface conductance at the coolant-entry face $x = \delta$ is a uniform h_c. The plate face $x = 0$ is exposed to a gas of uniform downstream temperature t_g, and at this coolant-exit face the unit surface conductance is a uniform h_g.

In accordance with the results of Weinbaum and Wheeler for porous metal walls (1), it is assumed that within the wall itself, $0 \leq x \leq \delta$, the coolant and solid temperatures are equal. Heat conduction within the wall is assumed to occur only in the solid material. A heat balance on a differential element of plate width dx identifies the conductive, convective and heat generation terms as $-k_w(1-p)dt/dx$, $-k_w(1-p)[dt/dx + (d^2t/dx^2)dx]$, $G_cC_c(dt/dx)dx$, and $q_w'''(1-p)dx$, and leads, by these assumptions, to the following differential equation to be satisfied by the plate temperature $t(X)$

$$\frac{d^2t}{dX^2} + a\frac{dt}{dX} + \lambda = 0 \quad \ldots\ldots [1]$$

with the abbreviations $a = G_cC_c\delta/(1-p)k_w$ and $\lambda = q_w'''\delta^2/k_w$, and the normalized coordinate $X = x/\delta$. Its general solution is

$$t = C_1 + C_2e^{-aX} - \frac{\lambda}{a}X \quad \ldots\ldots [2]$$

The boundary conditions on $t(X)$ are (Fig. 1)

$$\left.\begin{aligned} -k_w(1-p)\frac{1}{\delta}\left(\frac{dt}{dX}\right)_0 &= h_g(t_g - t_o) \\ G_cC_c(t_o - t_c) &= q_w'''\delta(1-p) + h_g(t_g - t_o) - h_c(t_\delta - t_c) \end{aligned}\right\} \quad [3]$$

where $t_o = t(X=0)$ and $t_\delta = t(X=1)$ are unknown if $h_c \neq \infty \neq h_g$. The first relation expresses continuity between the heat conducted away from the exit face $X = 0$ and the heat gain at this face from the ambient gas at t_g. The second relation is an extension of that given by Mayer and Bartas (3) which identifies the rate of enthalpy increase of the coolant as being equal to the rate of internal heat generation plus the difference in heat rate entering and leaving at the two exposed surfaces.

Local Wall Temperature. The two boundary conditions in Equation [3] determine the required form of the integration constants C_1 and C_2 in Equation [2] as

$$C_1 = \frac{(N_c+1)(N_g+1)\left(t_c + \frac{\lambda}{a}\right) - N_cN_g\left(t_g - \frac{\lambda}{a^2N_g}\right) \times e^{-a} + \frac{\lambda}{a^2}(N_g+1)}{(N_c+1)(N_g+1) - N_cN_ge^{-a}} \quad \ldots [4]$$

and

$$C_2 = \frac{\frac{\lambda}{a}(N_c+1)N_g\left(\frac{1}{M} - \frac{1}{aN_g} - 1\right) - \frac{\lambda}{a^2}N_g}{(N_c+1)(N_g+1) - N_cN_ge^{-a}} \quad \ldots [5]$$

where $M = (\lambda/a)/(t_g - t_c)$, $N_c = h_c/G_cC_c$, and $N_g = h_g/G_cC_c$. The value of M can be positive or negative according to whether t_g exceeds or is less than t_c; if $t_g = t_c$, then $M = \infty$. By means of Equations [2, 4, 5] the steady temperature t can be calculated at any plane in the plate.

Eight special cases are of interest. The case of negligible surface resistances is represented by $h_c = \infty = h_g$. Identifying

t_o as t_g and t_b as t_c and taking $N_c = \infty = N_g$ in Equations [4, 5] gives, according to Equation [2]

$$\text{I.}\quad \frac{t}{\lambda} = \frac{\left(\frac{t_b}{\lambda} + \frac{1}{a}\right)(1 - e^{-aX}) - \frac{t_o}{\lambda}(e^{-a} - e^{-aX})}{(1 - e^{-a})} - X \quad \ldots\ldots [6]$$

If h_g is finite while $h_c = 0$, then the temperature profile is given by

$$\text{II.}\quad \frac{t - t_c}{\lambda/a} = \left[\left(\frac{1}{M} - 1\right)\left(\frac{N_g}{N_g + 1}\right) - \frac{1}{a}\right]e^{-aX} - X + \frac{1}{a} + 1 \quad \ldots\ldots [7]$$

The surface temperatures $(t_o - t_c)/\lambda$ $(X = 0)$ and $(t_b - t_c)/\lambda$ $(X = 1)$ calculated by Equation [7] are shown in Fig. 2 as a function of the dimensionless parameters "a" and $A = (1/M - 1)N_g/(N_g + 1)$, only positive values of the latter being given. The value $A = 0$ corresponds either to Case IV ($N_g = 0$) or the condition $(t_g - t_c) = \lambda/a$. Positive values of A imply that $M < 1$ with $t_g > t_c$, while negative values occur if $M < 0$ or if $M > 1$ with $t_g > t_c$. For the practical case $t_g = t_c$, $A = -N_g/(N_g + 1)$. For $M = 1$, $A = 0$ and the solution becomes independent of heat transfer at the exit surface. The curves show how rapidly the entrance surface temperature $(X = 1)$ decreases with increasing coolant flow as compared to the exit surface temperature $(X = 0)$. For a given value of "a" the difference in surface temperatures increases the larger the value of A.

Case III is obtained from Equation [7] by setting $N_g/(N_g + 1) = 1$. Thus

$$\text{III.}\quad \frac{t - t_c}{\lambda/a} = \left(\frac{1}{M} - \frac{1}{a} - 1\right)e^{-aX} - X + \frac{1}{a} + 1 \quad \ldots\ldots [8]$$

In this case $M = (\lambda/a)/(t_o - t_c)$. Green's solution (6) follows directly from Equation [7] by taking $N_g = 0$ as

$$\text{IV.}\quad \frac{t - t_c}{\lambda/a} = \frac{1}{a}(1 - e^{-aX}) - X + 1 \quad \ldots\ldots [9]$$

Similar solutions for cases of zero internal heat generation $(q_w''' = 0)$ are obtained by taking $\lambda = 0$. This gives the solution of Mayer and Bartas (3) as[3]

$$\text{V.}\quad \frac{t}{t_g} = \frac{\left(1 - \frac{t_c}{t_g}\right)e^{-aX} + \left(1 + \frac{1}{N_g}\right)\frac{t_c}{t_g} - \frac{e^{-a}}{(1 + 1/N_c)}}{\left(1 + \frac{1}{N_g}\right) - \frac{e^{-a}}{(1 + 1/N_c)}} \quad \ldots\ldots [10]$$

If $h_c = \infty = h_g$ the solution is (3)

$$\text{VI.}\quad \frac{t}{t_o} = \frac{\left(1 - \frac{t_b}{t_o}\right)e^{-aX} - e^{-a} + \frac{t_b}{t_o}}{(1 - e^{-a})} \quad \ldots\ldots [11]$$

Further, if $h_c = 0$ (3, 5)

$$\text{VII.}\quad \frac{t - t_c}{t_g - t_c} = \frac{N_g}{N_g + 1}e^{-aX} \quad \ldots\ldots [12]$$

or if $h_c = 0$ and $h_g = \infty$ (4)

$$\text{VIII.}\quad \frac{t - t_c}{t_o - t_c} = e^{-aX} \quad \ldots\ldots [13]$$

[3] The well-known solution for $G_c = 0$

$$\frac{t - t_c}{t_g - t_c} = \frac{1 + N_c a(1 - X)}{1 + N_c a + N_c/N_g}$$

may be derived independently or taken directly from Equation [10].

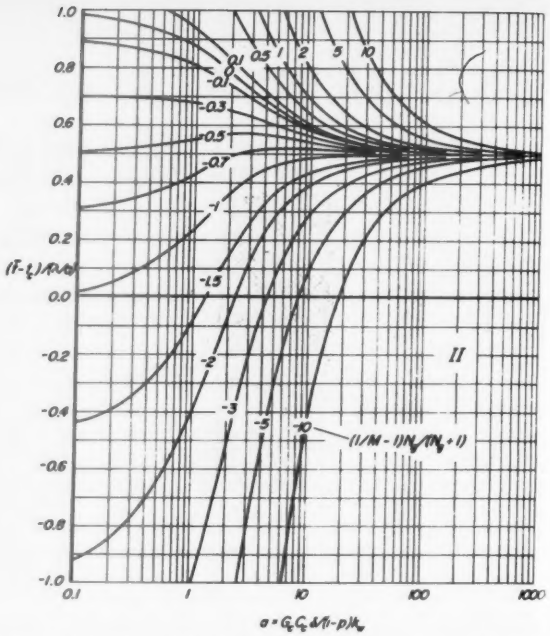

Fig. 3 Mean porous plate temperatures $\bar{t}$ for Case II ($h_c = 0$)

Mean Wall Temperature. The mean wall temperature is calculated from the temperature profile as

$$\bar{t} = \int_0^1 t\,dX \quad \ldots\ldots [14]$$

For the general case the mean temperature can be calculated from C_1 and C_2 in Equations [4, 5] as $\bar{t} = C_1 - C_2(e^{-a} - 1)/a - \lambda/2a$. Specific results for the eight special cases are

$$\text{I.}\quad \frac{\bar{t}}{\lambda} = \frac{\left(\frac{t_b}{\lambda} + \frac{1}{a}\right)(e^{-a} + a - 1) - \frac{t_o}{\lambda}[(1 + a)e^{-a} - 1]}{a(1 - e^{-a})} - \frac{1}{2} \quad \ldots\ldots [15]$$

$$\text{II.}\quad \frac{\bar{t} - t_c}{\lambda/a} = \left[\left(\frac{1}{M} - 1\right)\left(\frac{N_g}{N_g + 1}\right) - \frac{1}{a}\right] \times \frac{(1 - e^{-a})}{a} + \frac{1}{a} + \frac{1}{2} \quad \ldots\ldots [16]$$

$$\text{III.}\quad \frac{\bar{t} - t_c}{\lambda/a} = \left(\frac{1}{M} - \frac{1}{a} - 1\right)\frac{(1 - e^{-a})}{a} + \frac{1}{a} + \frac{1}{2} \quad \ldots\ldots [17]$$

$$\text{IV.}\quad \frac{\bar{t} - t_c}{\lambda/a} = \frac{1}{a^2}(e^{-a} + a - 1) + \frac{1}{2} \quad \ldots\ldots [18]$$

$$\text{V.}\quad \frac{\bar{t}}{t_g} = \frac{\left(1 - \frac{t_c}{t_g}\right)\frac{(1 - e^{-a})}{a} + \left(1 + \frac{1}{N_g}\right)\frac{t_c}{t_g} - \frac{e^{-a}}{(1 + 1/N_c)}}{\left(1 + \frac{1}{N_g}\right) - \frac{e^{-a}}{(1 + 1/N_c)}} \quad \ldots\ldots [19]$$

$$\text{VI.}\quad \frac{\bar{t}}{t_o} = \frac{\left(1 - \frac{t_b}{t_o}\right)\frac{(1 - e^{-a})}{a} - e^{-a} + \frac{t_b}{t_o}}{(1 - e^{-a})} \quad \ldots\ldots [20]$$

$$\text{VII.}\quad \frac{\bar{t} - t_c}{t_g - t_c} = \frac{N_g}{N_g + 1}\frac{(1 - e^{-a})}{a} \quad \ldots\ldots [21]$$

$$\text{VIII.}\quad \frac{\bar{t} - t_c}{t_o - t_c} = \frac{1}{a}(1 - e^{-a}) \quad \ldots\ldots [22]$$

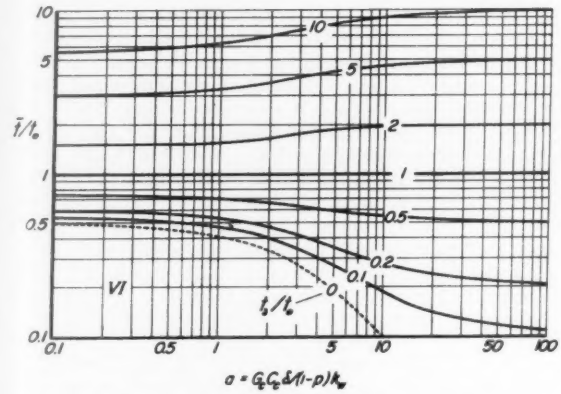

Fig. 4 Mean porous plate temperatures $\bar{t}$ for Case VI ($q_w''' = 0$, $h_c = \infty = h_g$)

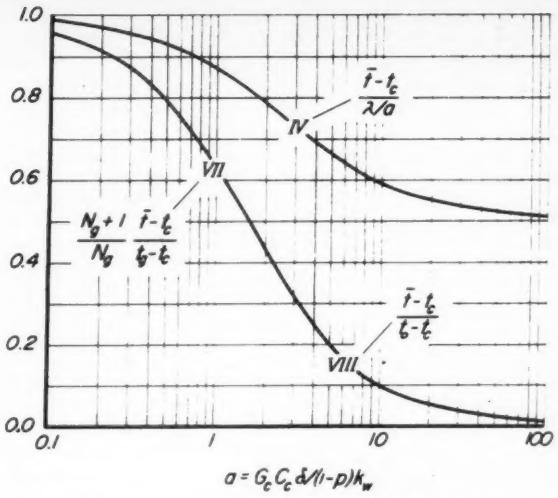

Fig. 5 Mean porous plate temperatures $\bar{t}$ for Case IV ($h_c = 0 = h_g$), Case VII ($q_w''' = 0 = h_c$) and Case VIII ($q_w''' = 0 = h_c, h_g = \infty$)

Equations [18, 22] are given in (6) and (4), respectively. Case II, Equation [16], is plotted in Fig. 3 for both positive and negative values of A. Case VI, Equation [20], is shown in Fig. 4, again with "a" as an independent parameter. As $a \to 0$ the temperature profile becomes linear and $\bar{t} \to \frac{1}{2}(t_o + t_\delta)$, while as $a \to \infty$, $\bar{t} \to t_\delta$. Surface temperature ratios for which $t_\delta/t_o \geq 1$ refer to the case where the porous wall is to be heated by a high temperature gas being forced through it. Fig. 5 gives the mean wall temperature for Cases IV, VII and VIII according to Equations [18, 21, 22]. The curve for Case IV is also contained in Fig. 3 ($A = 0$).

Efficiency and Surface Temperature Difference. The efficiency η_1 and surface temperature difference η_2, both normalized with respect to the over-all temperature difference $(t_g - t_c)$, are defined as

$$\eta_1 = \frac{t_g - t_o}{t_g - t_c} \qquad \eta_2 = \frac{t_o - t_\delta}{t_g - t_c} \quad \ldots\ldots [23, 24]$$

The general case is obtained as $\eta_1 = (t_g - C_1 - C_2)/(t_g - t_c)$ and $\eta_2 = [C_2(1 - e^{-a}) + \lambda/a]/(t_g - t_c)$, with C_1 and C_2 given by Equations [4, 5]. Thus

$$\eta_1 = \frac{1 + M\left[\left(\frac{N_c}{N_c+1}\right)\frac{(1-e^{-a})}{a} - 1\right]}{1 + N_g\left[1 - \left(\frac{N_c}{N_c+1}\right)e^{-a}\right]} \quad \ldots\ldots [25]$$

and

$$\eta_2 = \frac{\left[N_g - \frac{M}{a}\left(\frac{N_g}{N_c+1} + 1\right)\right](1-e^{-a}) + M\left(1 + \frac{N_g}{N_c+1}e^{-a}\right)}{1 + N_g\left(1 - \frac{N_c}{N_c+1}e^{-a}\right)} \quad \ldots\ldots [26]$$

Three special cases that have significance can be taken directly from Equations [25, 26] as

II. $$\eta_1 = \frac{1 - M}{N_c + 1}$$

$$\eta_2 = \frac{\left[N_g - \frac{M}{a}(N_g + 1)\right](1 - e^{-a}) + M(1 + N_g e^{-a})}{N_g + 1} \quad \ldots\ldots [27, 28]$$

V. $$\eta_1 = \frac{1}{1 + N_g\left(1 - \frac{N_c}{N_c+1}e^{-a}\right)}$$

$$\eta_2 = \frac{N_g(1 - e^{-a})}{1 + N_g\left(1 - \frac{N_c}{N_c+1}e^{-a}\right)} \quad \ldots\ldots [29, 30]$$

VII. $$\eta_1 = \frac{1}{N_g + 1}$$

$$\eta_2 = \frac{N_g}{N_g + 1}(1 - e^{-a}) \quad \ldots\ldots [31, 32]$$

Equation [27] shows that for $M > 1$ $(\lambda/a > t_g - t_c)$, $t_o > t_g$, while for $M < 1$, $t_o < t_g$. For $M = 1$, $t_o = t_g$. Equations [29, 30] are given in (3), and Equations [31, 32] are given in (3) and (5). For Cases V and VII, $\eta_1/\eta_2 = N_g(1 - e^{-a})$. The independence of this result on heat transfer at the coolant side (h_c) is significant only for Case V (3). An exit-surface temperature ratio may be defined which for Case VII becomes $(t_g - t_\infty)/(t_o - t_c) = 1/N_g$. This ratio was measured by Wheeler and Duwez (8) in experiments on transpiration cooling of a porous copper tube using hydrogen and nitrogen as coolants. As in η_1 for Case VII, this ratio is independent of δ and thus independent of geometry.

Thermal Stress. In the calculation of the steady-state thermal stresses it is assumed that the plate is free of external constraints and of end effects. It is also assumed that the usual stress relations that describe elastic behavior of solid materials apply in the present case of porous materials. This assumption will be less severe for plates of high solidity. Under these conditions the nonzero normal components of thermal stress are the same in the lateral and longitudinal directions y and z, $\sigma = \sigma_y = \sigma_z$, and equal to

$$\sigma^* = \frac{(1-\nu)}{E\beta}\sigma_{y,z} = 4\left(1 - \frac{3}{2}X\right)\int_0^1 t\,dX - t + 12\left(X - \frac{1}{2}\right)\int_0^1 tX dX \quad \ldots\ldots [33]$$

This is the same as the more familiar expression[4] written in

[4] See, for example, (9).

terms of a coordinate system whose origin is located at the midplane of the plate. In Equation [33], ν, E and β are the Poisson's ratio, modulus of elasticity and coefficient of linear thermal expansion for the plate material, each considered uniform. If σ is positive the stress is tensile, and if negative the stress is compressive. The first integral in Equation [33] is recognized as the mean plate temperature $\bar{t} = C_1 - C_2(e^{-a} - 1)/a - \lambda/2a$. The second integral can be calculated for the general case as $C_1/2 - (C_2/a^2)[(a + 1)e^{-a} - 1] - \lambda/3a$. The result for Case II, using Equations [7, 16], is

$$\text{II.} \quad \frac{\sigma^*}{\lambda B/a} = (12X + 4a - 6)\frac{(1 - e^{-a})}{a^2} - \frac{6}{a}[X - (1 - X)e^{-a}] - e^{-aX} \quad \ldots [34]$$

where $B = (1/M - 1)N_g/(N_g + 1) - 1/a = A - 1/a$. The right side of Equation [34] is seen to depend only on "a" and X; it is therefore independent of the various boundary conditions (9). For Case III, $B = (1/M - 1/a - 1)$; Case IV, $B = -1/a$; Case VII, $\lambda B = [a/(t_g - t_c)]N_g/(N_g + 1)$; and Case VIII, $\lambda B = a/(t_o - t_c)$. Thus

$$\text{III.} \quad \frac{\sigma^*}{\lambda B/a} = \frac{\sigma^*}{\frac{\lambda}{a}\left(\frac{1}{M} - \frac{1}{a} - 1\right)} \quad \ldots [35]$$

$$\text{IV.} \quad \frac{\sigma^*}{\lambda B/a} = \frac{-\sigma^*}{\lambda/a^2} \quad \ldots [36]$$

$$\text{VII.} \quad \frac{\sigma^*}{\lambda B/a} = \frac{N_g + 1}{N_g}\frac{\sigma^*}{(t_g - t_c)} \quad \ldots [37]$$

$$\text{VIII.} \quad \frac{\sigma^*}{\lambda B/a} = \frac{\sigma^*}{(t_o - t_c)} \quad \ldots [38]$$

Values of the maximum thermal stress for Cases II, IV, VII, and VIII are shown in Fig. 6. In each case the maximum stress occurs at the exit face $X = 0$ (subscript "o"), and for Cases II, VII, and VIII this maximum stress is compressive. Case VIII shows that the stress is zero for zero coolant flow and that it increases rapidly with increases in the coolant mass velocity G_c. Case IV is plotted as σ_o^*/λ so as to be independent of G_c. In this case the maximum stress is tensile; it vanishes for large values of "a" and approaches a limiting value of $\sigma_o^*/\lambda = \frac{1}{12}$ as "a" tends to zero. This limit occurs because in this limiting condition the surface temperatures approach an absolute difference of $\lambda/2$.

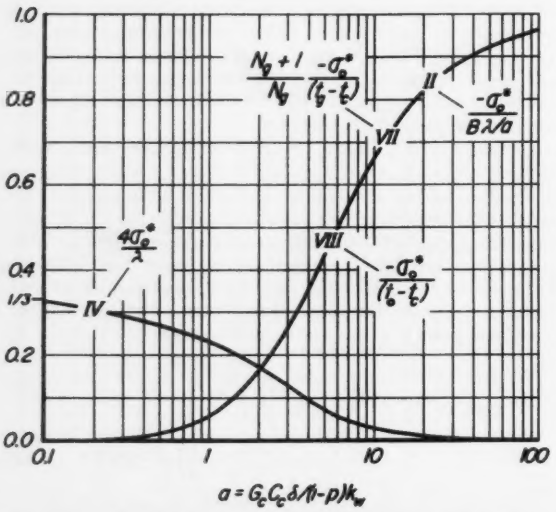

Fig. 6 Maximum porous plate thermal stresses σ^* for Case II ($h_c = 0$), Case IV ($h_c = 0 = h_g$), Case VII ($q_w''' = 0 = h_c$) and Case VIII ($q_w''' = 0 = h_c$, $h_g = \infty$)

Circular Tubes

General Solution. Consider a long circular tube as in Fig. 7 with an inner radius r_g and outer radius r_c, and with the coolant flowing inward through the tube wall. All other quantities are the same as defined in Fig. 1 with the exceptions that the wall thickness is now dependent on the surface radii and that the temperature of the outer surface is now denoted as t_1. Again the solid and coolant temperatures are assumed to be equal within the wall, $r_g \leq r \leq r_c$. Another difference with circular geometry is that the coolant mass velocity in the tube wall is not constant but depends on the local radius r. This dependence can be expressed in terms of a known mass velocity at either surface. Choosing the inner surface (3) and denoting the mass velocity here as G_{cg}, one has by mass continuity $G_{cg}2\pi r_g = (G_{cr}/p)2\pi r p$, or $G_{cr} = G_{cg}/R$, where R is the local radius normalized with respect to the inner radius as $R = r/r_g$.

Identifying the conductive, convective and generation components in an element of wall thickness dr as $-k_w 2\pi r(1 - p)dt/dr$, $-k_w 2\pi(r + dr)(1 - p)[dt/dr + (d^2t/dr^2)dr]$, $2\pi r_g G_{cg} C_c (dt/dr)dr$ and $q_w''' 2\pi r(1 - p)dr$ gives the characteristic differential equation to be satisfied by $t(R)$ as

$$\frac{d^2t}{dR^2} + \frac{(1 + b)}{R}\frac{dt}{dR} + \mu = 0 \quad \ldots [39]$$

Here $b = G_{cg}C_c r_g/(1 - p)k_w$ and $\mu = q_w''' r_g^2/k_w$, these quantities being analogous to "a" and λ in plane geometry. The general solution of Equation [39] is

$$t = C_1 + C_2 R^{-b} - \frac{\mu}{2(b + 2)} R^2 \quad \ldots [40]$$

The boundary conditions on $t(R)$ are (divided by 2π)

$$\left\{ \begin{aligned} -k_w(1 - p)\left(\frac{dt}{dR}\right)_1 &= h_g r_g (t_g - t_o) \\ G_{cg}C_c r_g (t_o - t_c) &= q_w''' \frac{(1 - p)}{2}(r_c^2 - r_g^2) + h_g r_g (t_g - t_o) - h_c r_c (t_1 - t_c) \end{aligned} \right\} \quad \ldots [41]$$

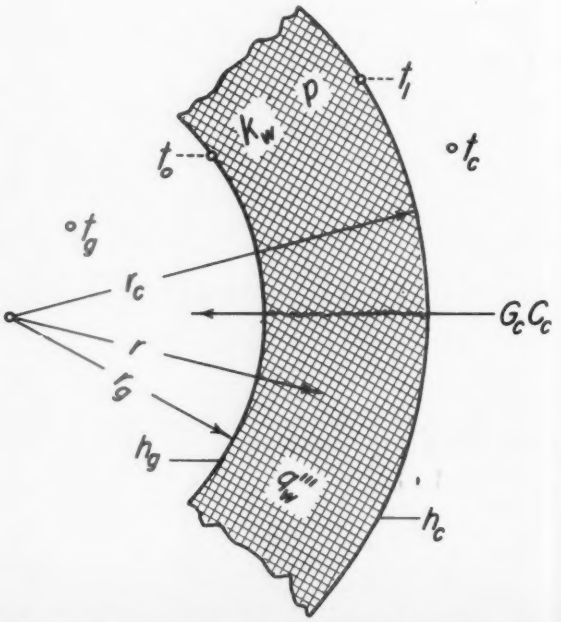

Fig. 7 Porous, heat-generating, transpiration-cooled tube

Local Wall Temperature. The integration constants C_1 and C_2 in Equation [40] are, according to the boundary conditions in Equation [41]

$$C_1 = \frac{(N_c R_c + 1)(N_g + 1)t_c - N_c N_g t_g R_c^{1-b} + \frac{\mu}{2b}(N_g + 1)(R_c^2 - 1) + \frac{\mu}{b(b+2)}[N_c R_c^{1-b} + (N_g + 1)] - \frac{\mu R_c^2}{2(b+2)}[N_c N_g R_c^{1-b} - (N_c R_c + R_c^{-2})(N_g + 1)]}{(N_c R_c + 1)(N_g + 1) - N_c N_g R_c^{1-b}} \quad [42]$$

and

$$C_2 = \frac{\frac{\mu}{b}\left\{\frac{N_c N_g b R_c}{2(b+2)}(1 - R_c^2) + (N_c R_c + 1)\frac{N_g}{P} + \frac{1}{2}N_g(1 - R_c^2) - \frac{1}{(b+2)}[N_c R_c + (N_g + 1)]\right\} r_g^b}{(N_c R_c + 1)(N_g + 1) - N_c N_g R_c^{1-b}} \quad [43]$$

where $R_c = r_c/r_g$ and $P = (\mu/b)/(t_g - t_c)$ which is analogous to M in plane geometry. For circular tubes the N_g and N_c become $N_g = h_g/G_{cg}C_c$ and $N_c = h_c/G_{cg}C_c$. By means of Equations [40, 42, 43] the steady temperature t can be calculated at any radius in the tube wall.

Solutions for the eight special cases are

I. $$\frac{t}{\mu} = \frac{\left[\frac{t_i}{\mu} + \frac{(R_c^2 - 1)}{2(b+2)}\right](1 - R^{-b}) - \frac{t_o}{\mu}(R_c^{-b} - R^{-b})}{(1 - R_c^{-b})} - \frac{(R^2 - 1)}{2(b+2)} \quad [44]$$

II. $$\frac{t - t_c}{\mu/b} = \left\{\left[\frac{1}{P} - \frac{1}{2}(R_c^2 - 1)\right]\frac{N_g}{N_g + 1} - \frac{1}{(2+b)}\right\} R^{-b} - \frac{bR^2}{2(b+2)} + \frac{R_c^2}{2} \quad [45]$$

III. $$\frac{t - t_c}{\mu/b} = \left[\frac{1}{P} - \frac{1}{2}(R_c^2 - 1) - \frac{1}{(2+b)}\right] R^{-b} - \frac{bR^2}{2(b+2)} + \frac{R_c^2}{2} \quad [46]$$

IV. $$\frac{t - t_c}{\mu/b} = \frac{R_c^2}{2} - \frac{\left(R^{-b} + \frac{b}{2}R^2\right)}{(b+2)} \quad [47]$$

V.[5] $$\frac{t}{t_g} = \frac{\left(1 - \frac{t_c}{t_g}\right)R^{-b} + \left(1 + \frac{1}{N_g}\right)\frac{t_c}{t_g} - \frac{R_c^{1-b}}{(R_c + 1/N_c)}}{\left(1 + \frac{1}{N_g}\right) - \frac{R_c^{1-b}}{(R_c + 1/N_c)}} \quad [48]$$

VI. $$\frac{t}{t_o} = \frac{(1 - t_i/t_o)R^{-b} - R_c^{-b} + t_i/t_o}{(1 - R_c^{-b})} \quad [49]$$

VII. $$\frac{t - t_c}{t_g - t_c} = \frac{N_g}{N_g + 1} R^{-b} \quad [50]$$

VIII. $$\frac{t - t_c}{t_o - t_c} = R^{-b} \quad [51]$$

[5] The familiar solution for $G_{cg} = 0$

$$\frac{t - t_o}{t_g - t_c} = \frac{1 + N_c b \ln(R_c/R)}{1 + N_c b \ln R_c + N_c R_c/N_g}$$

may be derived independently or taken directly from Equation [48].

Mean Wall Temperature. The mean wall temperature is calculated as

$$\bar{t} = \frac{2}{(R_c^2 - 1)}\int_1^{R_c} tR\,dR \quad [52]$$

The solution for the general case can be calculated from C_1 and C_2 in Equations [42, 43] as $\bar{t} = C_1 + 2C_2(R_c^{2-b} - 1)/(2 - b)(R_c^2 - 1) - \mu(R_c^2 + 1)/4(b + 2)$. Results for the eight special cases are

I. $$\frac{\bar{t}}{\mu} = \frac{\left[\frac{t_i}{\mu} + \frac{(R_c^2 - 1)}{2(b+2)}\right]\left[(R_c^2 - 1) + \frac{2}{(b-2)}(R_c^{2-b} - 1)\right] - \frac{t_o}{\mu}\left[R_c^{-b}(R_c^2 - 1) + \frac{2}{(b-2)}(R_c^{2-b} - 1)\right]}{(R_c^2 - 1)(1 - R_c^{-b})} \quad [53]$$

II. $$\frac{\bar{t} - t_c}{\mu/b} = \left\{\left[\frac{2}{P(1 - R_c^2)} + 1\right]\frac{N_g}{N_g + 1} - \frac{2}{(b+2)(1 - R_c^2)}\right\}\frac{(R_c^{2-b} - 1)}{(b-2)} - \frac{b(R_c^2 + 1)}{4(b+2)} + \frac{R_c^2}{2} \quad [54]$$

III. $$\frac{\bar{t} - t_c}{\mu/b} = \left[\frac{2}{P(1 - R_c^2)} - \frac{2}{(b+2)(1 - R_c^2)} + 1\right] \times \frac{(R_c^{2-b} - 1)}{(b-2)} - \frac{b(R_c^2 + 1)}{4(b+2)} + \frac{R_c^2}{2} \quad [55]$$

IV. $$\frac{\bar{t} - t_c}{\mu/b} = \frac{1}{(b+2)} \times \left[\frac{2(R_c^{2-b} - 1)}{(b-2)(R_c^2 - 1)} - \frac{b}{4}(R_c^2 + 1)\right] + \frac{R_c^2}{2} \quad [56]$$

V. $$\frac{\bar{t}}{t_g} = \frac{\frac{2}{(2-b)}(1 - t_c/t_g)(R_c^{2-b} - 1)/(R_c^2 - 1) + (1 + 1/N_g)t_c/t_g - R_c^{1-b}/(R_c + 1/N_c)}{(1 + 1/N_g) - R_c^{1-b}/(R_c + 1/N_c)} \quad [57]$$

VI. $$\frac{\bar{t}}{t_o} = \frac{\frac{2}{(b-2)}(1 - t_i/t_o)(R_c^{2-b} - 1)/(1 - R_c^2) - R_c^{-b} + t_i/t_o}{(1 - R_c^{-b})} \quad [58]$$

VII. $$\frac{\bar{t} - t_c}{t_g - t_c} = 2\frac{N_g}{N_g + 1}\frac{(R_c^{2-b} - 1)}{(b-2)(1 - R_c^2)} \quad [59]$$

VIII. $$\frac{\bar{t} - t_o}{t_o - t_c} = 2\frac{(R_c^{2-b} - 1)}{(b-2)(1 - R_c^2)} \quad [60]$$

Case VI, Equation [58], is shown in Fig. 8 for two values of the radius ratio $R_c = 1.25$ and 1.50. As $b \to 0$, $\bar{t} \to t_o - (t_i - t_o)[\frac{1}{2}\ln R_c - R_c^2/(R_c^2 - 1)]$, and for $b \to \infty$, $\bar{t} \to t_i$. The curves suggest that the range $0.1 < b < 1$ is less significant than the same range in the case of plane geometry (Fig. 4). Cases IV, VII and VIII are plotted in Fig. 9 for radius ratios $R_c = 1.25$, 1.50, 1.75 and 2. For Case IV as $b \to 0$, $\bar{t} \to t_c + (\mu/2b)(R_c^2 - 1)$, and as $b \to \infty$, $\bar{t} \to t_c + (\mu/4b)(R_c^2 - 1)$.

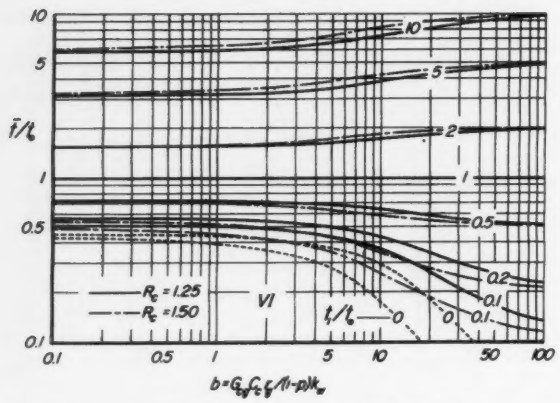

Fig. 8 Mean porous tube temperatures $\bar{t}$ for Case VI ($q_w''' = 0$, $h_c = \infty = h_g$)

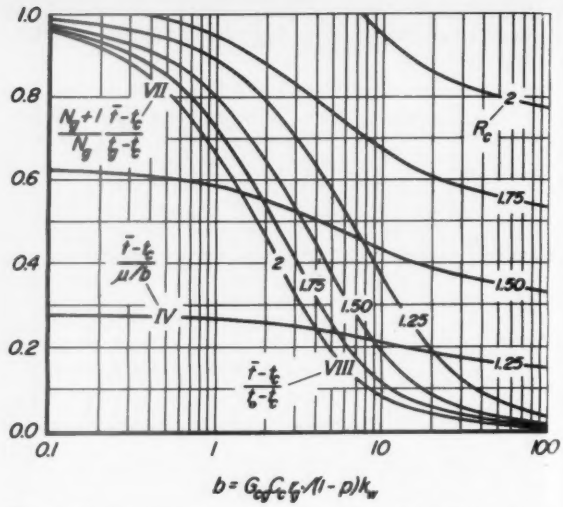

Fig. 9 Mean porous tube temperatures $\bar{t}$ for Case IV ($h_c = 0 = h_g$), Case VII ($q_w''' = 0 = h_c$), and Case VIII ($q_w''' = 0 = h_c$, $h_g = \infty$)

Efficiency and Surface Temperature Difference. The efficiency and surface temperature difference defined by Equations [23, 24] are, by Equations [40, 42, 43]

$$\eta_1 = \frac{1 + P\left[\dfrac{(N_c R_c b + b + 2)(1 - R_c^2) - 2N_c R_c(R_c^{-b} - 1)}{2(b+2)(N_c R_c + 1)}\right]}{1 + N_g\left[1 - \dfrac{R_c^{1-b}}{(R_c + 1/N_c)}\right]} \quad \ldots\ldots [61]$$

and

$$\eta_2 = \frac{\left\{N_g - \dfrac{P}{2(b+2)(N_c R_c + 1)}\,[2(N_c R_c + N_g + 1) + N_g(N_c R_c b - b - 2)(1 - R_c^2)]\right\}(1 - R_c^{-b}) + \dfrac{Pb(1 - R_c^2)}{2(b+2)}\left\{1 + N_g\left[1 - \dfrac{R_c^{1-b}}{(R_c + 1/N_c)}\right]\right\}}{1 + N_g\left[1 - \dfrac{R_c^{1-b}}{(R_c + 1/N_c)}\right]} \quad \ldots [62]$$

where for the tube $\eta_2 = (t_o - t_i)/(t_g - t_c)$. Results for three special cases are contained in Equations [61, 62] as

II. $$\eta_1 = \frac{1 - \dfrac{P}{2}(R_c^2 - 1)}{N_g + 1}$$

$$\eta_2 = \left\{\frac{N_g}{N_g + 1}\left[1 + \frac{P}{2}(1 - R_c^2)\right] - \frac{P}{(b+2)}\right\} \times (1 - R_c^{-b}) + \frac{Pb(1 - R_c^2)}{2(b+2)} \quad \ldots [63, 64]$$

V. $$\eta_1 = \frac{1}{1 + N_g\left[1 - \dfrac{R_c^{1-b}}{(R_c + 1/N_c)}\right]}$$

$$\eta_2 = \frac{N_g(1 - R_c^{-b})}{1 + N_g\left[1 - \dfrac{R_c^{1-b}}{(R_c + 1/N_c)}\right]} \quad \ldots\ldots [65, 66]$$

VII. $$\eta_1 = \frac{1}{N_g + 1}$$

$$\eta_2 = \frac{N_g}{N_g + 1}(1 - R_c^{-b}) \quad \ldots\ldots [67, 68]$$

For Cases V and VII, $\eta_1/\eta_2 = N_g(1 - R_c^{-b})$.

Thermal Stress. General equations for the radial, tangential (circumferential) and longitudinal components of thermal stress in a long tube free of external constraints and of end effects are given by (10)

$$\sigma_r^* = \frac{1}{R^2}\left(\frac{R^2 - 1}{R_c^2 - 1}\int_1^{R_c} tR\,dR - \int_1^R tR\,dR\right) \quad \ldots [69]$$

$$\sigma_t^* = \frac{1}{R^2}\left(\frac{R^2 + 1}{R_c^2 - 1}\int_1^{R_c} tR\,dR + \int_1^R tR\,dR\right) - t \quad \ldots [70]$$

$$\sigma_z^* = \frac{2}{(R_c^2 - 1)}\int_1^{R_c} tR\,dR - t \quad \ldots\ldots [71]$$

Results for the general case can be obtained from Equation [40] using C_1 and C_2 from Equations [42, 43]. Specific results for the three special cases II, IV and VII are

II. $$\frac{\sigma_r^*}{\mu/b} = \frac{1}{(2-b)R^2}\left[\frac{R^2 - 1}{R_c^2 - 1}(R_c^{2-b} - 1) - (R^{2-b} - 1)\right] \times \left\{\left[\frac{1}{P} - \frac{1}{2}(R_c^2 - 1)\right]\frac{N_g}{N_g + 1} - \frac{1}{(b+2)}\right\} + \frac{b}{8(b+2)R^2}(R^2 - 1)(R^2 - R_c^2) \quad \ldots [72]$$

II. $$\frac{\sigma_t^*}{\mu/b} = \frac{1}{(2-b)R^2}\left[\frac{R^2 + 1}{R_c^2 - 1}(R_c^{2-b} - 1) + (b-1)R^{2-b} - 1\right] \times \left\{\left[\frac{1}{P} - \frac{1}{2}(R_c^2 - 1)\right]\frac{N_g}{N_g + 1} - \frac{1}{(b+2)}\right\} + \frac{b}{8(b+2)R^2}[1 + 3R^4 - (R^2 + 1)(R_c^2 + 1)] \quad \ldots [73]$$

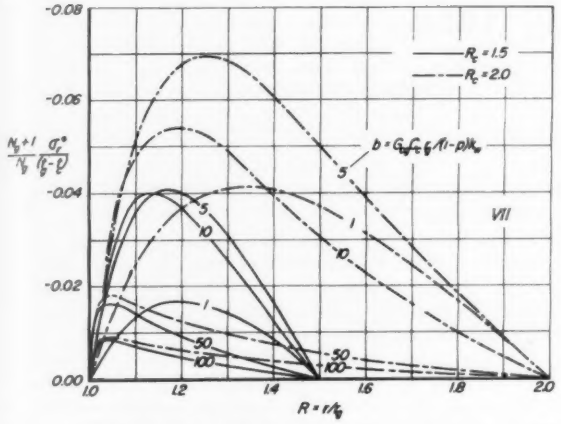

Fig. 10 Radial porous tube thermal stresses σ_r^* for Case VII ($q_w''' = 0 = h_c$)

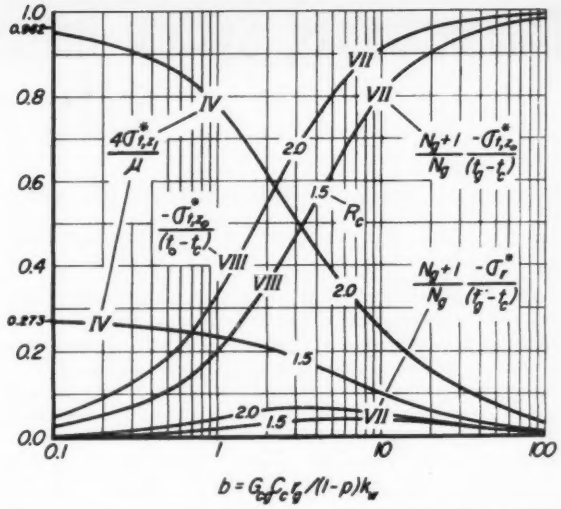

Fig. 11 Maximum porous tube radial, tangential and longitudinal thermal stresses σ_r^*, σ_t^* and σ_z^* for Case IV ($h_c = 0 = h_g$), Case VII ($q_w''' = 0 = h_c$), and Case VIII ($q_w''' = 0 = h_c$, $h_g = \infty$)

II. $$\frac{\sigma_z^*}{\mu/b} = \frac{1}{(2-b)}\left[2\left(\frac{R_c^{2-b}-1}{R_c^2-1}\right) + (b-2)R^{-b}\right]\times \left\{\left[\frac{1}{P} - \frac{1}{2}(R_c^2-1)\right]\frac{N_g}{N_g+1} - \frac{1}{(b+2)}\right\} + \frac{b}{4(b+2)}(2R^2 - R_c^2 - 1) \quad \ldots [74]$$

IV. $$\frac{\sigma_r^*}{\mu/b} = \frac{(R^{2-b}-1) - (R^2-1)(R_c^{2-b}-1)/(R_c^2-1) - b(b-2)(R^2-1)(R^2-R_c^2)/8}{(2+b)(2-b)R^2} \quad \ldots [75]$$

IV. $$\frac{\sigma_t^*}{\mu/b} = \frac{1-(b-1)R^{2-b} - (R^2+1)(R_c^{2-b}-1)/(R_c^2-1) - b(b-2)[1+3R^4 - (R^2+1)(R_c^2+1)]/8}{(2+b)(2-b)R^2} \quad \ldots [76]$$

IV. $$\frac{\sigma_z^*}{\mu/b} = \frac{(2-b)R^{-b} - 2(R_c^{2-b}-1)/(R_c^2-1) - b(b-2)(2R^2-R_c^2-1)/4}{(2+b)(2-b)} \quad \ldots [77]$$

VII. $$\frac{N_g+1}{N_g}\frac{\sigma_r^*}{(t_g-t_c)} = \frac{1}{(2-b)R^2}\times [(R^2-1)(R_c^{2-b}-1)/(R_c^2-1) - (R^{2-b}-1)] \quad \ldots [78]$$

VII. $$\frac{N_g+1}{N_g}\frac{\sigma_t^*}{(t_g-t_c)} = \frac{1}{(2-b)R^2}[(R^2+1)(R_c^{2-b}-1)/(R_c^2-1) + (b-1)R^{2-b} - 1] \quad \ldots [79]$$

VII. $$\frac{N_g+1}{N_g}\frac{\sigma_z^*}{(t_g-t_c)} = \frac{2}{(2-b)}\times (R_c^{2-b}-1)/(R_c^2-1) - R^{-b} \quad \ldots [80]$$

The radial thermal stresses σ_r^* for Case VII are shown in Fig. 10. These stresses are compressive and they vanish at the surfaces $R = 1$ and $R = R_c$ for all values of b and at all radii for both $b = 0$ and $b = \infty$. Hence, for some finite value of b the radial stress is a maximum at some radius $1 < R < R_c$. The maximum radial stress for a given value of b occurs, by Equation [78], at the radius $[(2R_c^2/b)(1 - R_c^{-b})/(R_c^2 - 1)]^{1/(2-b)}$. The radial stresses for Case IV are negligibly small. Results of calculations on the maximum tangential and longitudinal stresses for Case VII are given in Fig. 11 for two radius ratios $R_c = 1.5$ and 2.0. The behavior can be compared with the plate stresses in Fig. 6. Although not shown, the radial stresses for Case VIII are the same except that in this case $-[(N_g + 1)/N_g]\sigma_r^*/(t_g - t_c)$ is to be interpreted as $-\sigma_r^*/(t_g - t_c)$.

The tangential and longitudinal stresses for Case IV are equal and tensile. The maximum stresses in this case, plotted as σ_1^*/μ which is independent of G_{cg}, occur at the outer surface $R = R_c$ (subscript "1"). As $b \to 0$ these maximum surface stresses approach a maximum value equal, by Equations [76 or 77], to $\ln R_c/2(R_c^2 - 1) + (R_c^2 - 1)/8 - \frac{1}{4}$. For Cases VII and VIII the tangential and longitudinal stresses are again equal if one considers only the maximum values. However, in these cases the maximum stresses are compressive and they occur at the inner surface $R = 1$ (subscript "o"). The effect of radius ratio R_c appears to be most pronounced in Case IV. The reason for this is suggested in Fig. 9.

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Demonstrating System Reliability by the Sequential Probability Ratio Test

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The Sequential Probability Ratio Test is presented as a practical statistical approach to demonstrating the reliability requirements of rocket systems and subsystems. In this test, the sample size is not predetermined but, rather, the testing is continued and the results analyzed after each system is tested, until the results are sufficient to indicate a decision. The main advantages of this procedure are (a) a generally smaller sample size than that required by other statistical procedures, (b) all calculations can be done prior to testing and the test data can be recorded graphically without additional computations.

It is shown that the required calculations can be reduced to determining the equations of two straight lines. These two straight lines are then plotted on graph paper and the test results are recorded by a plot line on the graph. Testing is terminated as soon as this plot line intersects either of the straight lines. If the upper line is intersected, the system has not demonstrated adequate reliability and rejection is indicated. If the lower line is intersected, the system has met the reliability requirements and acceptance is indicated.

Nomenclature

R_0 = reliability desired by the consumer
R_1 = minimum reliability acceptable to the consumer; $R_0 > R_1$
B = consumer's risk
A = producer's risk
gm = number of successes that have occurred thus far in the testing
dm = number of failures that have occurred thus far in the testing
$P_1 = 1 - R_1$
$P_0 = 1 - R_0$
log = natural logarithm

A ROCKET reliability demonstration program must first show that each component will function reliably as a part of the system. Secondly, the reliability of the complete system must be demonstrated.

Component reliability demonstration techniques have been developed by Robert Lusser[3] and have been widely accepted. However, there is no widely accepted technique for demonstrating the reliability of the integrated complex subsystems and systems. For this reason, the Sequential Probability Ratio Test is presented herein.

Advantages of a Sequential Procedure

A sequential procedure is a type of statistical evaluation program in which the number of systems to be tested is not predetermined but, rather, the testing is continued, and the results analyzed after each system is tested, until the results are sufficient to indicate a decision.

According to the older statistical procedures, the sample size would be determined before the testing is begun. But even though the results may be conclusive early in the testing, because the data are not evaluated after each system is tested, it would go unnoticed and all the following systems are literally wasted.

Missile systems are expensive and usually testing is, by necessity, destructive. It is obvious that any missile statistical program should test only enough, and not more, systems than are necessary to give the required results. This is the big advantage of a sequential procedure.

The second advantage is that all calculations can be done prior to testing and they are very simple. It will be shown that there are actually no computations to be made once testing is begun and this is in spite of the results being analyzed after each system is tested.

As an example, sample size for the classical statistical approach is 300 but seven out of the first 20 units tested have failed. Sequential analysis may indicate rejection right at this point. But 280 additional units would be tested and wasted if the older statistical procedure were used.

Sequential Probability Ratio Test

Whenever the testing results for each system can be expressed as go or no go, good or bad, satisfactory or unsatisfactory, and the universe of the type of system under test can be considered infinite or at least very large, there would be a certain percentage of good systems in the universe of that type of item. The purpose of the testing is to determine if this percentage of good systems is satisfactory.

By testing one system at a time the percentage of successes will tend to come closer and closer to the true percentage of good systems in the universe. The universe of a system can be thought of as being the total number of systems of the same type that will be produced.

The SPR test shows for every sample size, beyond the minimum number of required consecutive successes or failures, an acceptance number and a rejection number. This means that for any size sample, if the number of successes is greater than or equal to the acceptance number, testing may be terminated and the type of system accepted as having the required percentage of good systems in its universe (i.e., adequate reliability). If the number of systems tested successfully is below or equal to the rejection number, the type of system can immediately be rejected for inadequate reliability.

It is apparent that since a decision must be made on the basis of a sample there is always the possibility of sampling error. A satisfactory universe may be rejected or an unsatisfactory universe may be accepted. Therefore the customer must prescribe a risk, B, which he is willing to take of accepting a bad universe and the producer must prescribe a risk, A, which he is willing to take of rejecting a good universe. Also the customer requires at least a certain fraction, R_0, of good items in an accepted universe, but will not be hurt too much if the true reliability is slightly lower, R_1.

The SPR test, then, consists of testing, one unit at a time, until the results indicate rejection at A risk to the producer or acceptance at B risk to the customer.

When the SPR Test May Be Applied

The SPR test is applicable when:

1 The parties concerned can agree on two reliability figures, $R_0 > R_1$, such that if the true reliability, of the type of

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unit being tested, is less than R_1, the error resulting from acceptance is of great practical consequence. And, the error resulting from rejection is of great practical consequence if the true reliability is equal or greater than R_0. If the true reliability is between R_1 and R_0, no particular harm would result from a decision of either acceptance or rejection.

2 The universe from which the sample is taken is infinite or, at least, very large. This means that the systems tested must be as similar as possible to those that are to be produced in the future.

3 A consumer's risk (B) and a producer's risk (A) are chosen.

4 The existing systems can be randomly sampled and the field conditions adequately simulated by the test conditions.

How to Apply the SPR Test

This paper will now show how to apply the SPR test equations. Derivations will not be given but they can be found in Reference (6).

Procedure

The equations and relationships used in the SPR test can be reduced to two straight lines determined by

$$gm \log \frac{R_1}{R_0} + dm \log \frac{P_1}{P_0} = \log \frac{1 - B}{A} \quad \ldots\ldots\ldots [1]$$

$$gm \log \frac{R_1}{R_0} + dm \log \frac{P_1}{P_0} = \log \frac{B}{1 - A} \quad \ldots\ldots\ldots [2]$$

Number of failures (i.e., dm) is plotted on the vertical axis and the number of successes (i.e., gm) is plotted on the horizontal axis.

The two straight lines are plotted by substituting the assigned values of R_1, R_0, A and B into the two equations and graphing the result. This is done prior to commencement of testing. All that remains, once testing has begun, is to run a plot line one unit to the right after a success or one unit up after a failure. Testing is continued and the plot line is extended either a unit to the right (for a success) or a unit up (for a failure) until the plot line intersects one of the two straight lines. If the upper line is intersected, rejection is indicated. If the lower line is intersected, acceptance of the system as having the required reliability is indicated.

Example 1

$R_0 = 0.98$ $\quad$ $R_1 = 0.94$
$A = 0.05$ $\quad$ $B = 0.05$

By substituting these values into the two equations, two parallel straight lines are obtained and then plotted as shown in Fig. 1. The ordinate represents the number of failures and the abscissa represents the number of successes. As the testing proceeds the successful and unsuccessful tests are plotted in sequence.

Testing is terminated when: (a) The lower line is intersected; the reliability of the system is considered acceptable. The consumer's specifications have been realized. Or (b) the upper line is intersected; the reliability of the system is then considered unacceptable and rejection follows. The consumer's specifications have not been attained.

One of the main advantages of the sequential approach is that the true reliability value will usually favor either R_0 or R_1 and the sequential test will quickly accept the value favored. Another advantage is that it is not necessary to be concerned with variance or calculations of the data since the data are completely recorded by extending the plot line until it intersects either of the straight lines.

In this figure, it is seen that 70 consecutive successful tests are sufficient to accept the system and only three initial failures are required for rejection. Therefore, if the first three tests are failures, the testing can be terminated with rejection of the system, indicating that redesign and/or improvement is necessary.

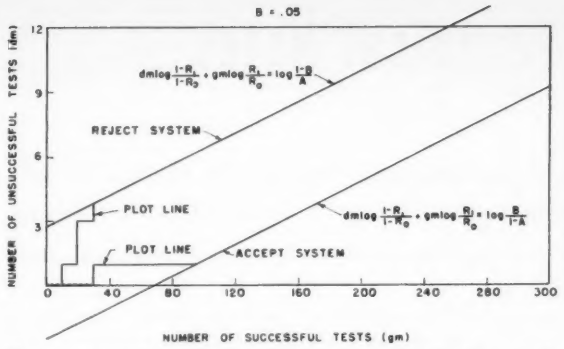

Fig. 1 Sequential probability ratio for $R_0 = 0.98$, $R_1 = 0.94$, $A = 0.05$, $B = 0.05$

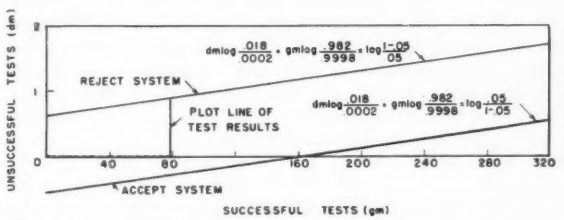

Fig. 2 Sequential probability ratio for $R_0 = 0.9998$, $R_1 = 0.982$, $A = 0.05$, $B = 0.05$

In this figure two situations are given:

In one situation, 30 consecutive successful tests (gm) were recorded, then one failure (dm) followed and then 65 consecutive successful tests—a total of 96 tests. On the last test the lower line was intersected, resulting in a decision to accept.

In the other case, the test results gave the following sequence: 10 successes, 1 failure, 10 successes, 2 failures, 10 successes, 1 failure. This total of 34 tests resulted in the top line being intersected on the last test, indicating rejection.

Example 2

$R_0 = 0.9998$ $\quad$ $R_1 = 0.982$
$A = 0.05$ $\quad$ $B = 0.05$

By substituting these values into the two equations, two parallel straight lines are obtained and then plotted as shown in Fig. 2.

The plot line in Fig. 2 represents the following sequence: 79 successes, 1 failure. This total of 80 tests resulted in the top line being intersected on the last test, indicating rejection.

Expected Sample Size

It is shown in the references that the actual expected sample size cannot be determined a priori because the true reliability (i.e., percentage of successes in the universe) is unknown. However, by assuming various values for reliability, the average sample number curve can be plotted. The largest expected sample size occurs for reliability values between R_0 and R_1. Even in this, the worst case, the expected sample size is usually practical.

RMI has had good results by assuming the true reliability is equal to R_0 and using this value to calculate average sample size.

The average sample number (ASN) equation is

$$\text{ASN} = \frac{(1 - A) \log \dfrac{B}{1 - A} + A \log \dfrac{1 - B}{A}}{(1 - R_0) \log \dfrac{1 - R_1}{1 - R_0} + R_0 \log \dfrac{R_1}{R_0}}$$

For Example 1, the ASN is 138. *(Continued on page 899)*

Technical Notes

Spectroscopic Studies of the Effect of Boundary Layer Flow on Flameholding[1]

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Daniel and Florence Guggenheim Jet Propulsion Center, Princeton University, Princeton, N. J.

A spectral intensity method for measuring the local fuel-air ratio in two-dimensional hydrocarbon-air flames is used to explore the phenomenon of preferential diffusion in the wake of bluff body flameholders. The findings support the hypothesis that turbulence in the boundary layer and in the adjacent flow reduces the degree of preferential diffusion and this, in turn, explains the observed shift in flameholder stability limits.

Nomenclature

CH/C_2	= ratio of spectral radiations at 4300 Å to spectral radiations at 5150 Å (band width approx. 100 Å)
L	= length of recirculation zone
Re	= Reynolds number based on mainstream velocity and flameholder diameter
t_{chem}	= chemical time characteristic of stabilization mechanism
$U_{B.O.}$	= blow-off velocity
x	= axial distance from ℄ of flameholder
$\frac{[CH/C_2]_l}{[CH/C_2]_0}$	= radiation deviation parameter
ϕ_F	= fuel equivalence ratio

Subscripts

0 = mainstream
l = local

Introduction

IN CURRENT theories of the mechanism of flameholding by bluff bodies in high speed streams of fuel-air mixtures, the state of the boundary layer as it separates from the flameholder has an important effect on the blow-off curves (stability limits) (1, 2).[5] Whether the boundary layer is laminar or turbulent determines the extent of preferential diffusion of fuel and oxygen into the stabilizing eddy; this in turn affects the blow-off velocities. It is the purpose of this note to report on the application of a previously described spectral intensity technique (3) to the exploration of the extent of such preferential diffusion as it is affected by the condition of the boundary layer and the adjacent flow.

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[1] This note is based on part of the research performed by the two junior authors in fulfillment of the requirements for advanced degrees.

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[3] Graduate Student in Aeronautics, candidate for the Ph.D. degree. Now with ARDE Associates, Newark, N. J. Mem. ARS.

[4] Professor of Jet Propulsion, Department of Aeronautical Engineering. Mem. ARS.

[5] Numbers in parentheses indicate References at end of paper.

Spectral Intensity Method

The visible radiation in premixed hydrocarbon-air flames is due for the most part to the emitters CH, C_2 and CO_2*. Propitiously, the intensity ratio of the radiation emitted at 4300 Å to that emitted at 5150 Å (denoted hereafter as CH/C_2) in a laminar flame is a unique function of fuel/air ratio (see 3, Fig. 1). The curve of intensity ratios for a laminar flame can be used (within limits) as a calibration curve to determine the local fuel/air ratio in any section of a two-dimensional flame of nonuniform mixture ratio burning at approximately the same temperature and pressure. For convenient reference in this paper, the calibration curve from (3) for a propane-air laminar flame has been replotted in Fig. 1. It is useful in the presentation of the intensity ratio traverses to normalize the ratio by dividing the value at each station along the traverse by the asymptotic value at the end of the traverse. Thus, the normalized intensity ratio $[CH/C_2]_l/[CH/C_2]_0$ is plotted in Figs 2, 3, and 4. $[CH/C_2]_l$ refers to the local radiation intensity ratio in the nonuniform reaction zone under investigation; $[CH/C_2)_0$ refers to mainstream conditions. $[CH/C_2]_0$ is determined by the measured intensity ratio downstream of the flameholding region where preferential diffusion effects (see below) are seen to have disappeared. Then, from $[CH/C_2)_l/[CH/C_2]_0$ it is possible to estimate immediately the deviation of the local fuel equivalence ratio from the mainstream value. Since the radiation parameter $[CH/C_2]_l/[CH/C_2]_0$ can be measured in an arbitrarily small zone of the flameholding region, depending only on the sensitivity of the receiver, the method permits detailed study of the progress of diffusion. This is to be compared with sampling methods that usually produce only gross average measurements of composition in the entire zone.

The spectral method for the determination of local fuel/air ratios in a reaction zone is, of course, dependent on the assumption that the burning conditions in the reaction zone under investigation are identical to those in the laminar calibration flame. Ideally, this assumption can be checked by measurement of the local fuel/air ratio using another experimental technique, e.g., chemical sampling. This direct comparison was outside the scope of the present program. However, an indirect check on the method was obtained from the observation that the absolute value of $[CH/C_2]_0$ measured locally in the mainstream was always within ±10 per cent of the value of $[CH/C_2]$ obtained from the calibration curve (at an identical mixture ratio).

The process of selective diffusion in the wake of a flame stabilizer reveals itself only at low Reynolds number (1, 2). In the Reynolds number regime ($Re < 10^4$) the mechanism of stabilization is influenced by the preferential molecular diffusion of fuel and oxygen into the flameholder region. Since, in general, the fuel and oxidizer can be expected to have different diffusivities, reaction in the flameholder region occurs at a different fuel/air ratio than that in the mainstream.

Experimental Results

The flow system (propane-air) and the radiation detection system (photomultiplier tube-interference filters) have been described in (4). As in the earlier experiments, combustion

EDITOR'S NOTE: This section of JET PROPULSION is open to short manuscripts describing new developments or offering comments on papers previously published. Such manuscripts are published without editorial review, usually within two months of the date of receipt. Requirements as to style are the same as for regular contributions (see masthead page of this issue).

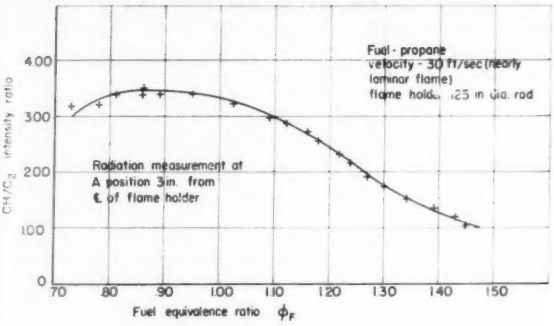
Fuel - propane
velocity - 30 ft/sec (nearly laminar flame)
flame holder .125 in dia. rod
Radiation measurement at
A position 3in. from
℄ of flame holder
CH/C2 intensity ratio
Fuel equivalence ratio ϕ_F

Fig. 1 CH/C_2 vs. fuel equivalence ratio at a station remote from the flameholder (calibration curve, reproduced from Ref. 3)

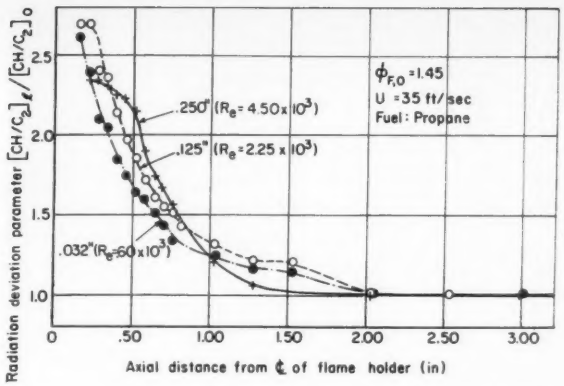
$\phi_{F,0}$ = 1.45
U = 35 ft/sec
Fuel: Propane
.250" (R_e = 4.50 x 10^3)
.125" (R_e = 2.25 x 10^3)
.032" (R_e = 560 x 10^3)
Radiation deviation parameter $[CH/C_2]_l/[CH/C_2]_0$
Axial distance from ℄ of flame holder (in)

Fig. 3 $[CH/C_2]_l/[CH/C_2]_0$ vs. distance downstream from flameholder—effect of flameholder diameter

took place in a $^1/_2$-in. × 2-in. two-dimensional duct, 12 in. long with Vycor glass walls. The flameholders were mounted parallel to the short side of the duct. The distribution of $(CH/C_2)_l/[CH/C_2]_0$ was determined downstream of the stabilizer as a function of (a) boundary layer disturbance, (b) mainstream turbulence, (c) flameholder diameter and (d) stabilizer Reynolds number.

The general view of the equipment is shown in Fig. 5. The radiation detection assembly, including photomultiplier, interference filter and collimator (for eliminating stray radiation), was mounted on a universal feed mechanism with a horizontal and vertical motion. During operation, the burner and the detection assembly were surrounded completely with a blackened enclosure. (One panel of the enclosure was removed to obtain the photograph in Fig. 5.) The pressure taps shown along the outer wall of the burner were not used in the present program. Typical flames anchored on two rod-type flameholders, i.e., $^1/_8$ and $^1/_2$ in. diam are presented in Fig. 6. In the experiments reported herein, measurements were confined to the "ears" of the flame immediately downstream of the stabilizer. Fig. 7 shows a flame stabilized on two different flat plate ($4^1/_2$ in. long and $^1/_8$ in. thick) holders. In this case, both flat plates have identical nominal cross sections; however, the plate on the right has been artificially roughened with boundary layer trips. It is noted that in either case the flameholding region appears to be similar to that characteristic of the simple rod.

Inspection of the results shown in Fig. 2 shows that strong deviations from the free stream radiation intensity ratio occur near the stabilizer, and that this deviation gradually disappears at greater distances downstream of the stabilizer. This gradual decay of the intensity ratio deviation is indicative of the gradual disappearance of preferential diffusion effects (see Fig. 1). Fig. 2 also shows that the insertion of an $^1/_8$-in. mesh turbulence grid upstream of the stabilizer resulted in a downward shift of the $[CH/C_2]_l/[CH/C_2]_0$ distribution. The presence of local turbulence (introduced by boundary layer trips) caused an even greater downward shift of $[CH/C_2]_l/[CH/C_2]_0$. Thus, boundary layer turbulence produced either by mainstream or local disturbances results in a reduction in the degree of selective diffusion.

In Fig. 3 a comparison is made of $[CH/C_2]_l/[CH/C_2]_0$ at constant flow conditions for three flameholders of different diameters. Contrary to expectations, the distributions indicate that the extent of the selective diffusion zone is nearly independent of flameholder diameter. In dimensionless distances, the region of selective diffusion for the 0.032-in. stabilizer extends more than fifty rod diameters, while for the 0.25 in. stabilizer it extends only six or seven rod diameters. Thus, at least for small rods, it is probable that the preferential diffusion process continues far downstream of the termination of the recirculation zone.

The data presented in Figs. 3 and 4 show that preferential diffusion occurs in all tests in the range of Reynolds numbers from 0.5×10^3 to 10×10^3. This finding agrees with the observation (1) that the blow-off curves for this type of air-fuel

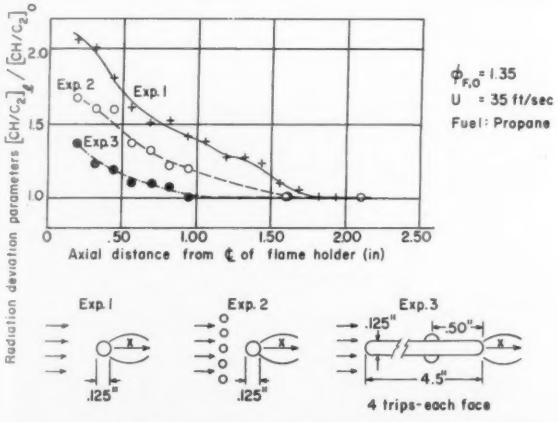

Fig. 2 $[CH/C_2]_l/[CH/C_2]_0$ vs. distance downstream from flameholder—effect of local and mainstream turbulence

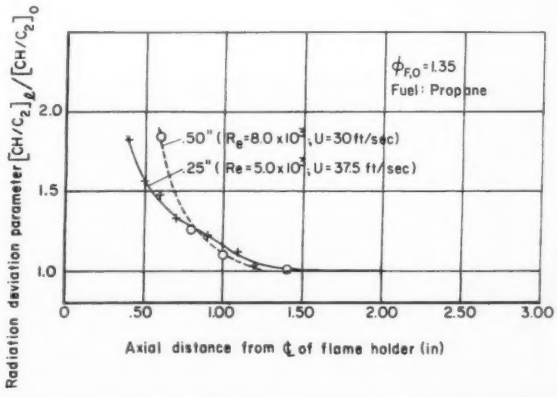
$\phi_{F,0}$ = 1.35
Fuel: Propane
.50" (R_e = 8.0 x 10^3; U = 30 ft/sec)
.25" (R_e = 5.0 x 10^3; U = 37.5 ft/sec)
Radiation deviation parameter $[CH/C_2]_l/[CH/C_2]_0$
Axial distance from ℄ of flame holder (in)

Fig. 4 $[CH/C_2]_l/[CH/C_2]_0$ vs. distance downstream from flameholder—effect of rod Reynolds number

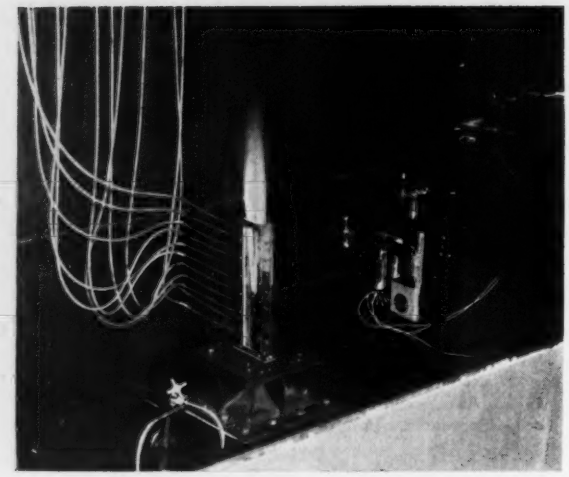

Fig. 5 Experimental apparatus with enclosed burner on left and radiation detection assembly at right

mixture are shifted to the rich side of the stoichiometric value. The flow capacity of this equipment was inadequate to perform tests above 10^4 Reynolds number.

Discussion

It has been recently suggested (5) that the process of blow-off from a bluff body stabilizer, immersed in premixed combustibles, can be expressed in the form

$$U_{B.O.} = L/t_{chem}$$

The chemical and fluid dynamical aspects of the stabilization mechanism are separated into the independent parameters t_{chem} and L; L is the length of the recirculation zone and t_{chem} can be interpreted as the induction time for ignition of the freestream. If the time $(L/U_{B.O.})$ during which fresh combustibles are in contact with the hot wake is less than the time for ignition (t_{chem}) the flame will not anchor. The existence of the preferential diffusion process and the resultant shifts in the stability limit curves imply that below the critical Reynolds number, $t_{chem,1}$ characteristic of the stabilization mechanism is not the same as $t_{chem,0}$ characteristic of the free stream. The significant chemical reaction time for blow-off is $t_{chem,1}$ and not $t_{chem,0}$.

To conclude, an important aspect of the experiments is the demonstration that the state of the boundary layer at the point of separation and in the stream adjacent to the stabilizing eddy has a significant effect on the radiation in this region. Although these experiments did not include blow-off tests, the correlations found previously between the radiation data and blow-off data of others suggests that the state of the flow in the boundary layer affects the blow-off limits through its effect on the local diffusional processes

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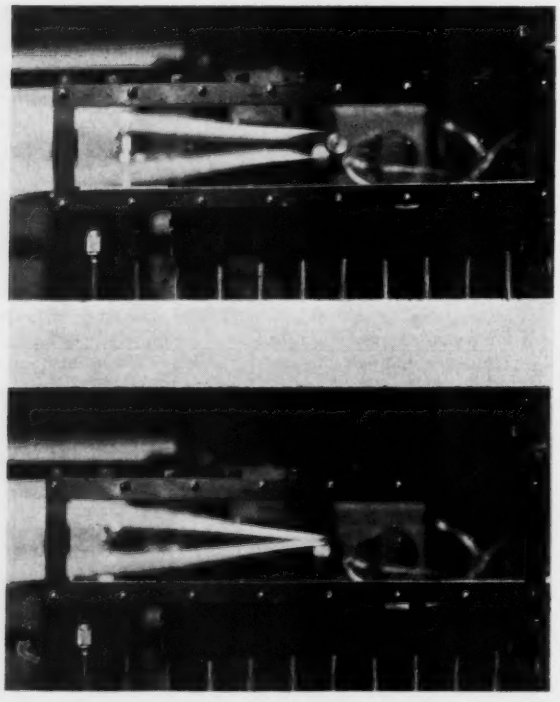

Fig. 6 Propane-air flames stabilized on cylindrical flameholders—¼-in. rod (below), ½-in. rod (above); combustible velocity 40 fps; equivalence ratio 1.40

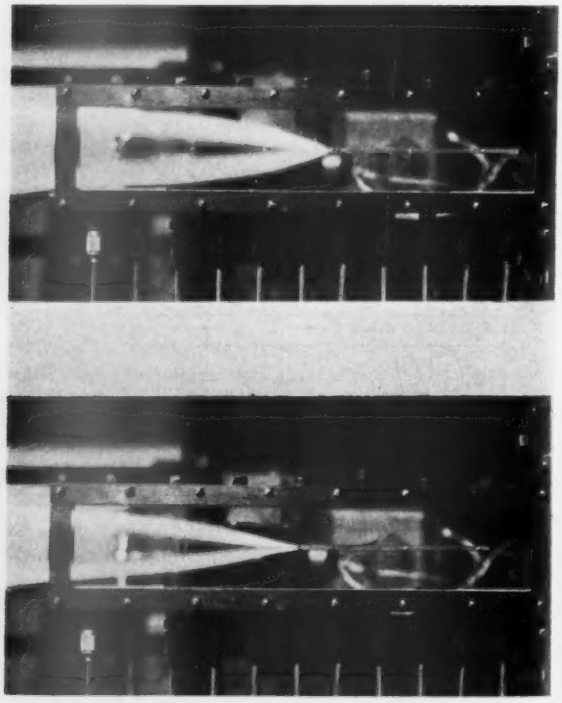

Fig. 7 Propane-air flames stabilized on flat plate flameholders—Flat plate (below), flat plate with boundary layer trips (above); combustible velocity 40 fps; equivalence ratio 1.40

Vectored Injection Into a Hypersonic Laminar Boundary Layer

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THE heat transfer in hypersonic flow has been investigated (1)[3] for vectored area injection of fluid into the boundary layer. Vectored injection corresponds to (a) transpiration cooling with injection oblique to the surface and (b) film cooling with evaporation.

Upon utilizing a modified Levy transformation (2), the boundary layer equations, in the absence of gas phase reactions, become

$$f_{\eta\eta\eta} + ff_{\eta\eta} = \tfrac{1}{2}(f_\eta^2 - 1 - s) \quad [1]$$

$$s_{\eta\eta} + fs_\eta = 0 \quad [2]$$

$$Z_{j_{\eta\eta}} + fZ_{j_\eta} = 0 \quad [3]$$

at the stagnation point of a blunt body.

In Equations [1–3] the Prandtl and Schmidt numbers were taken equal to unity and $\rho\mu$ was taken constant. The symbols are

C_f = skin friction coefficient $\tau_i/\frac{1}{2}\rho_i u_e^2$
C_h = Stanton number $h/\rho_i c_{p_i} u_e$
c_j = mass fraction of species j
c_p = specific heat at constant pressure
f_η = u/u_e
h = $K_i c_{p_i}\left(\frac{\partial T}{\partial y}\right)_i (H_e - H_i)^{-1}$
H = stagnation enthalpy
K = coefficient of thermal conductivity
s = $H/H_e - 1$
T = temperature
u, v = tangential and normal components of velocity
x, y, r_0 = coordinate system
Z_j = $s_i\left[1 - \frac{c_{j_i} - c_j(\eta)}{c_{j_i} - c_{j_e}}\right]$
η, ξ = similarity variables
ρ = density
μ = viscosity

subscripts

e = outer edge of boundary layer
i = inner edge of boundary layer
s = stagnation point

The solutions to Equations [1, 2, 3] were obtained on a REAC differential analyzer. It is noted that the complete seventh order system was treated as a fifth order system since with the particular choice of variables listed, Equations [2, 3] have precisely the same boundary conditions and hence are equivalent, that is

$$\eta = 0: \quad f_{\eta_i} = \frac{u_i}{u_e}, \qquad f_i = -v_i\left[2\left(\frac{du_e}{dx}\right)_s \frac{\mu_i}{\rho_i}\right]^{-\frac{1}{2}}$$

$$Z_{j_i} = s_i = \frac{H_i}{H_e} - 1 \quad [4]$$

$$\eta = \infty: \quad \lim_{\eta \to \infty} f_\eta = 1, \quad \lim_{\eta \to \infty} s = \lim_{\eta \to \infty} Z_j = 0$$

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[2] Research Engineer, Aerophysics Laboratory, Missile and Ordnance Systems Department.

[3] Numbers in parentheses indicate References at end of paper.

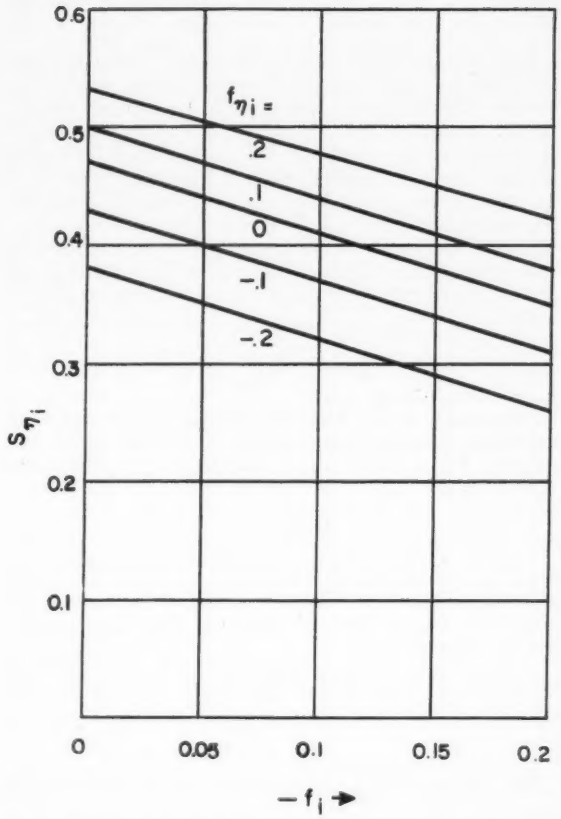

Fig. 1 Variation of enthalpy gradient with normal and tangential velocities

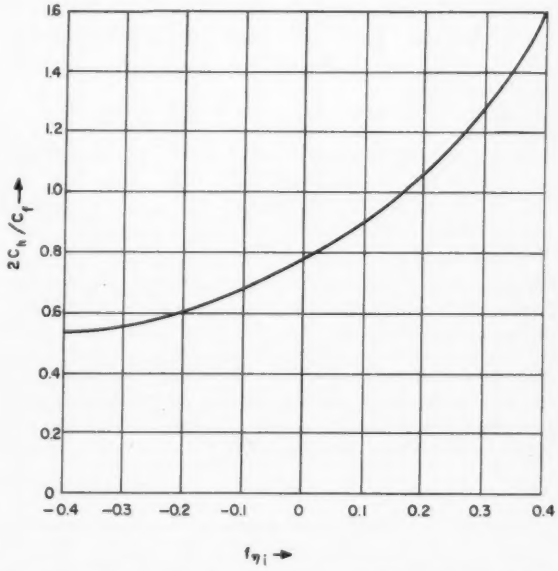

Fig. 2 Variation of ratio of Stanton number to skin friction coefficient with tangential velocity

A significant portion of the aerodynamic heat transfer at a stagnation point is due to conduction, which is given by

$$q_i = K_i \left(\frac{\partial T}{\partial y}\right)_i = \mu_i \left[2\left(\frac{du_s}{dx}\right)_s \frac{\rho_i}{\mu_i}\right]^{\frac{1}{2}} H_e s_{\eta i} \ldots\ldots [5]$$

It has been previously observed by many investigators that the effect of mass addition is to reduce the energy transfer to a surface. This result is indicated in Fig. 1, as the reduction in $s_{\eta i}$ with increasing $-f_i$. However, it is seen that the tangential component of velocity at the inner edge of the boundary layer also has a marked effect on the energy transfer. A tangential component in the direction of the main stream increases the heat transfer to the surface and vice versa.

The effect of vectored injection on skin friction was also investigated. The results may be expressed in terms of the ratio of Stanton number to skin friction coefficient, which becomes

$$\frac{2C_h}{C_f} = \frac{s_{\eta i}}{f_{\eta \eta i}} \ldots\ldots [6]$$

This ratio is weakly dependent on f_i up to values of -0.2, but has a strong dependence on the tangential velocity, as is shown in Fig. 2. Note that Reynold's analogy would predict that $2C_h/C_f = 1.0$.

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On Secondary Atomization of Droplets

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THE experimental work of Scheubel (1),[2] Klüsener (2), Siestrunck (3), Littaye (4), Lane (5) and Ryley (6) reveals that the criterion for secondary atomization (shattering of droplets) on collision can be represented by

$$\frac{\rho v^2 r}{\sigma} > \text{const} \ldots\ldots [1]$$

where

ρ = density of the liquid
σ = surface tension of liquid
v = velocity of impinging droplets
r = radius of droplets

This fact can be understood if we suppose that in a direct impact at fairly high velocities the two impinging droplets form roughly a disk of radius R and thickness ϵR. We may further assume that when $\epsilon < \epsilon_0 \ll 1$, i.e., when the radius of the disk is much larger than its thickness, the disk breaks into smaller droplets whose diameters are of the same order as the thickness of the disk. If $\epsilon > \epsilon_0$, the disk will attain a spherical form, resulting in the coalescence of droplets. Thus the condition for secondary atomization is

$$\epsilon < \epsilon_0 \ldots\ldots [2]$$

Conservation of energy and volume of liquid are expressed by

$$\sigma(2\pi R^2 + 2\pi \epsilon R^2 - 8\pi r^2) = 2 \cdot \frac{1}{2} \cdot \frac{4}{3} \pi \gamma^3 \rho v^2$$

$$2 \cdot \frac{4}{3} \pi r^3 = \pi \epsilon R^2$$

Hence we obtain

$$\frac{\rho r v^2}{\sigma} = 6\left\{\frac{(1+\epsilon)}{(3\epsilon)^{2/3}} - 1\right\} \ldots\ldots [3]$$

Since the right-hand side of Equation [3] is a decreasing function of ϵ in the range $0 < \epsilon < 2$, Equation [1] follows from Equations [2 and 3].

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A Mechanical Igniter for Liquid Rocket Engines[1]

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USUALLY rocket thrust chambers are ignited in three ways: (a) By pyrotechnic squibs for single start chambers, (b) by hypergolic or catalytic reaction, (c) by electric spark or glow plugs.

Only the glow or spark plug igniters have a ready multi-start capability. The spark plugs do not work well with acid-hydrocarbon systems because of quenching of the spark by the conductive acid. Neither do these igniters work well for high altitude starts.

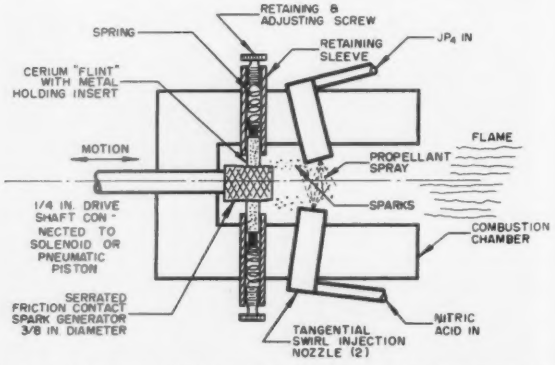

Fig. 1. Idealized sketch of the experimental igniter assembly

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[1] Department of Physics.

[2] Numbers in parentheses indicate References at end of paper.

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[1] The work for this paper was done at Bell Aircraft Corp. in 1955.

[2] Propulsion Head, Project Vanguard. Director, ARS.

[3] Rocket Engineer, Project Vanguard.

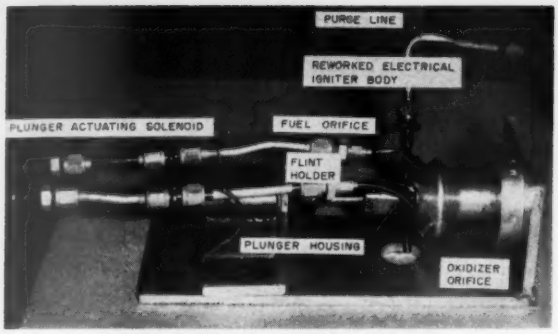

Fig. 2 Working model of igniter

The mechanical method, here, is based on the flint-spark principle. It was hoped with this to by-pass some of the disadvantages of the afore-mentioned "normal" ignition schemes. Millions of cigarette and gas lighters have been manufactured which are based on the flint spark principle. A hard steel ratchet or serrated wheel rubs a ferrocerium alloy (flint) and generates a shower of sparks which in turn ignite a gas or a wick saturated with some inflammable liquid.

A flint igniter was assembled by taking apart a rugged welding spark striker. Some of the parts were assembled as shown in Figs. 1 and 2. A serrated silica iron plunger was actuated by a solenoid; when a toggle switch was flipped the plunger rubbed on the flints and threw a great shower of sparks into the cavity of the little electrical igniter body.

RFNA and JP4 were injected through atomizing nozzles into the igniter chamber just at the moment when the sparks were generated. An acid lead was used.

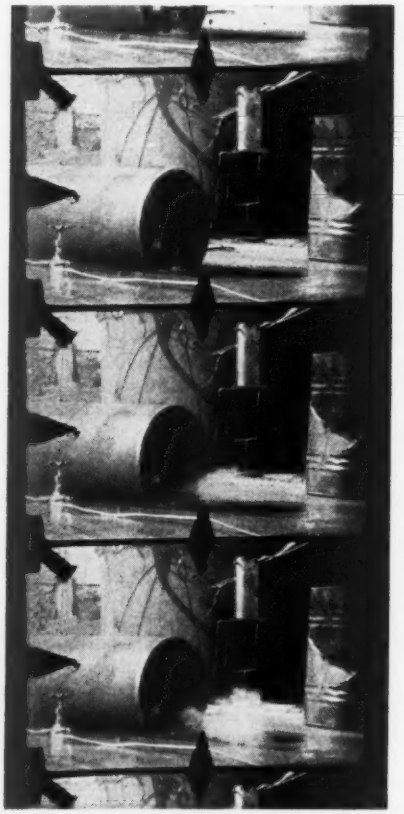

Fig. 3 Propellant ignition starting sequence (Fastax camera, 1500 frames/sec)

After several earlier development problems, mainly the proper sequencing of propellants and the construction of a spark generator, reliable ignition was obtained. High speed photographs and other measurements showed that no dangerous ignition delay occurred. The flints were not bothered by the acid; indeed, several commercial flints were soaked in WFNA and RFNA for several days. They were coated with a white substance which rubbed off on the first pass of the iron striker and did not prevent spark formation.

Several ignition tests were made at low temperatures (30 F) and at simulated high altitudes (about 40,000 ft). In all cases acid JP-4 ignition was obtained (Fig. 3).

Two tests were made with n-propyl nitrate at low temperatures and high altitudes. Again, ignition resulted.

It is believed that the usefulness of this ignition scheme results from the nature of the sparks which act as loci or little individual ignition sources. These little igniters, so to speak, travel through the vaporized propellant zone adjacent to each spark. One of the evils of high altitude ignition of propellants (whether in a rocket thrust chamber or ramjet) is the rapid transport of combustibles away from an ignition source, such as a spark or glow plug. The sparks, on the other hand, travel through much of the vaporized or gaseous propellant region and ignite in many local spots at once.

The spark generation does not drop off much with altitude. A shower of sparks was even obtained when the plunger and flints were actuated in a closed chamber filled with nitrogen or evacuated to the equivalent of 50,000 ft.

Little information exists on the nature of the "pyrophoric" action of these cerium alloys. The meager literature that exists states that the spark is really a small bit of ferro-cerium alloy oxidized to incandescence after being rubbed off. It is the authors' opinion that while oxidation is probably responsible for some of the spark, frictional incandescence of the metal also plays a part since sparks were produced in an environment having only a small fraction of sea level oxygen.

R. Paris and J. Zolch of the Ignition Branch, Wright Air Development Center, were most helpful with advice and encouragement.

The Determination of Thermal Entrance Lengths for Gases in Turbulent Flow in Smooth Round Ducts[1]

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THIS note describes the results of some recent experiments for determining the thermal entrance lengths for air and carbon dioxide in turbulent flow through a tube under conditions of large temperature difference between the flowing gas and the wall of the tube. In the experiments the ratio of the average wall to average gas bulk temperature, $\overline{T}_w/\overline{T}_b$, was varied from 2.11 to 2.72 for bulk Reynolds numbers ranging from approximately 50,000 to 250,000. The experiments were conducted in three different electrically heated test sections 0.569 in. ID having lengths of 21, 40 and 60 diameters; the test sections were made from Haynes Alloy 25(L605). In all

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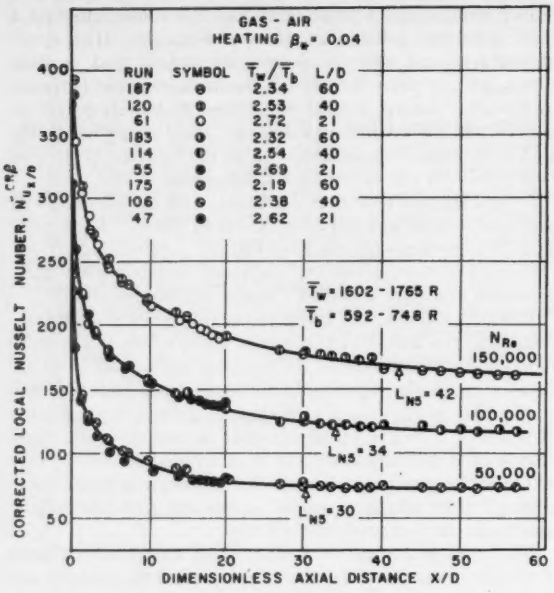

Fig. 1 Variation of corrected local Nusselt number with x/d for air; uniform initial temperature profile, fully developed initial velocity profile, constant heat flux

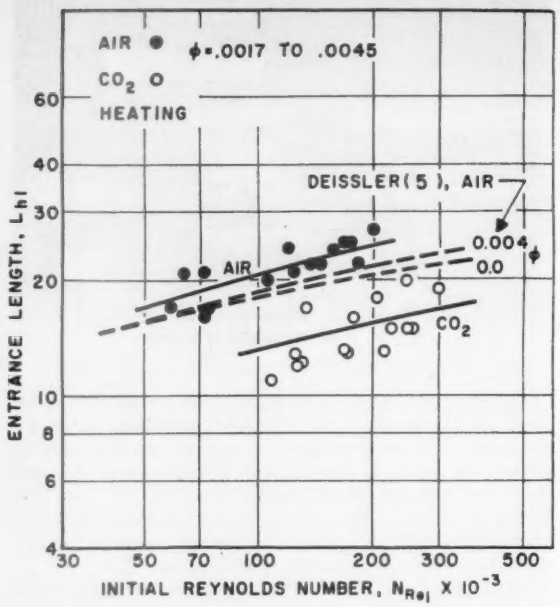

Fig. 3 Influence of initial Reynolds number upon one per cent entrance length; uniform heat flux, uniform initial temperature profile, fully developed initial velocity profile

cases the flow conditions at the entrance to the test section were characterized by a fully developed turbulent velocity profile and a uniform temperature profile.

The thermal entrance length is defined as the axial distance from the onset of energy exchange to the cross section where the thermal boundary layers meet in the center of the tube. The thermal entrance lengths reported in this note were inferred from measurements of the local heat transfer coefficient $h_{x/D}$ and from calculated values of the local Nusselt number $N_{Nu(x/D)}$ with the physical properties of the gas evaluated at the local bulk temperature. Because it is difficult to determine precisely the point at which the local heat transfer coefficient (or Nusselt number), when plotted as a function of x/D, becomes asymptotic to the x/D axis, it is convenient to define a "5 per cent" and a "1 per cent" entrance length L_{h5}, L_{h1} or L_{N5}, L_{N1} as the distance required for the local value of h or N_{Nu}, respectively, to attain a value equal to 1.05 or 1.01 times the asymptotic value.

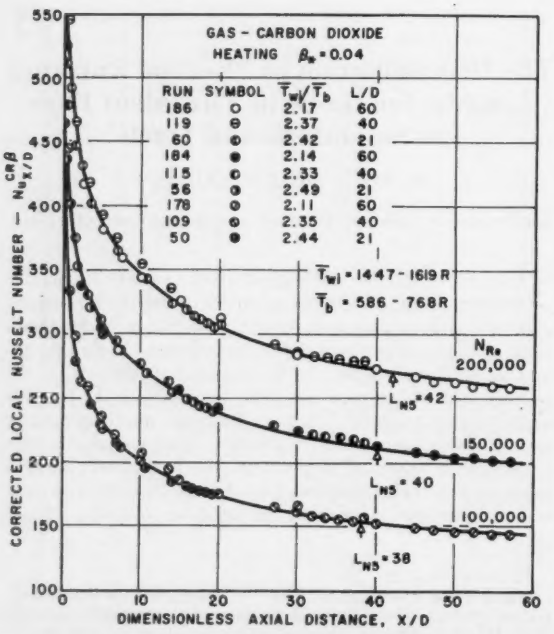

Fig. 2 Variation of local Nusselt number with x/d for carbon dioxide; uniform initial temperature profile, fully developed initial velocity profile, constant heat flux

The entrance lengths reported in the literature (1–4)[4] for turbulent flow are of the order of 10 diameters or less for fluids having a Prandtl number greater than 0.70. Reference (4) indicates that the influence of Reynolds number upon the entrance length for water, Prandtl number 6.5 to 8.0, was very small. Analytical predictions of the entrance length (6–10) based upon the calculated values of local Nusselt number assuming constant properties for the fluid and constant wall temperature indicate values less than 10 to 12 diameters for a fluid having a Prandtl number of 0.72, when the Reynolds number is 100,000. The analytical work of Deissler (5) which considers variable physical properties and uniform heat flux, predicts the thermal entrance lengths, based upon the calculated thickness of the thermal boundary layer, to be of the order of 16 to 20 diameters over an initial Reynolds number (viscosity evaluated at the temperature of the gas at the entrance to the test section) range of 50,000 to 150,000 for the case where the wall heats the gas.

Fig. 1 presents experimental values for the corrected local Nusselt number as a function of the axial distance, x/D, along the tube for air, for bulk Reynolds numbers of 50,000, 100,000 and 150,000. Fig. 2 presents similar data for carbon dioxide. The results presented in Figs. 1 and 2 were obtained with the three different test sections referred to earlier. It was not possible to maintain identical flow conditions in all three test sections; furthermore, the heat transfer parameter β does not remain constant along the length of the test section for the

[4] Numbers in parentheses indicate References at end of paper.

uniform heat flux boundary condition corresponding to electrical heating of the test section. It was necessary, therefore, that the local values of the Nusselt number be corrected to a nominal value of β and Reynolds number. In general, the heat transfer parameter β is defined (11) by the equation

$$\beta = \frac{q_w \sqrt{\tau_w/\rho_w}}{c_p g \tau_w t_w}$$

q_w = unit heat flux at the wall, Btu/sec ft²
τ_w = shear stress at the wall, lb/ft²
g = acceleration due to gravity, ft/sec²
ρ_w = mass density of the fluid at the wall, lb sec²/ft⁴
t_w = temperature at the wall, degrees Rankine

It is seen from the data and curves of Figs. 1 and 2 that the local values of the Nusselt number are not influenced appreciably by the length of the test section. The large values of the 5 per cent entrance length L_{N5}, which range from 30 to 42, arise from the influence of increasing bulk temperature upon the value of the thermal conductivity of the flowing gas and consequently upon the Nusselt number. The local value of the Nusselt number, therefore, does not attain a constant value but continually decreases with the axial distance along the electrically heated tube. For that reason the local Nusselt number is not a reliable criterion for establishing the value of the thermal entrance length for fluids having physical properties which vary appreciably with the temperature.

Fig. 3 presents the relationship of the 1 per cent entrance length L_{h1} for air and carbon dioxide as a function of the initial Reynolds number. The values of L_{h1} were obtained from the variation of the local heat transfer coefficient with axial distance.

Since the value of β is not constant along the length of the test section, Deissler (5) recommends the parameter ϕ defined as

$$\phi = \frac{q_w}{c_p \gamma_i \bar{u}_i t_i}$$

q_w = unit heat flux at the wall, Btu/sec ft²
c_p = specific heat at constant pressure, Btu/lb R
γ_i = specific weight at the initial temperature of the gas, lb/ft³
$\bar{u}_i$ = average velocity of fluid at the inlet to the test section, fps
t_i = inlet temperature, degrees Rankine

The results presented in Fig. 3 are correlated with the parameter ϕ. Fig. 3 compares the experimental values with those predicted by (5) from calculations of the thermal boundary layer thickness for the case where air is the heat transfer fluid. It is seen that the agreement is quite satisfactory. In the case where the flowing gas was carbon dioxide the experiments show that the entrance lengths L_{h1} are significantly less than those obtained for air for the same values of initial Reynolds number.

Figs. 1, 2 and 3 present results obtained from experiments where the gases were heated by the walls of the tube. Experiments are currently being conducted that will permit calculation of local heat transfer characteristics and consequent evaluation of the thermal entrance length for the case where hot gases are cooled by the walls of the tube.

The authors wish to express their appreciation to M. J. Zucrow for his helpful guidance during the course of the investigation and to the staff of the Jet Propulsion Center, Purdue University, for their assistance in constructing and maintaining the apparatus.

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Demonstrating System Reliability by SPR Test

(*Continued from page 891*)

Repeated Testing of the Same Systems

Ideally, a new system should be used for each test. However, this is usually economically unfeasible. Therefore, system testing must be limited to repeated testing of the same systems. This repeated testing of the same systems proves that if all components are reliable, the system will be reliable. But, component reliability has already been demonstrated by extensive test to failure. Therefore, repeated testing of the same systems is philosophically consistent as well as practical.

Conclusion

When values for the four parameters can be agreed upon, sequential analyses may be used. R_0 and a slightly lower (or compromise) figure, R_1, then, are determined, and testing is continued, as has been shown, onl until there is sufficient evidence to indicate rejection at A r to the producer or acceptance at B risk to the consumer.

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Solid Propellant Bibliography

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American Rocket Co., Wyandotte, Mich.

Introduction

THIS work was begun as a private compilation of all available entries in the field of solid propellants. However, the growing list indicated a vast total of unclassified literature which warranted a systematic search and subsequent publication. It was found that such a bibliography did not exist and that its existence would be a valuable research tool to all. At this point, Project SQUID agreed to sponsor the task.

The purpose of this bibliography is (1) to secure an over-all view of the state of the art, and (2) to implement and supplement the classified works available through the Solid Propellant Information Agency. This unclassified work should be of value to those in both the classified and unclassified fields. To the former it will indicate what has been done in an effort to avoid duplication. To the latter this work will find value as an introduction or a starting point for research.

As this bibliography reveals, the solid propellant field is rich in lore. The span represented here is roughly the years 1850–1956, and nearly 2000 entries are presented. Standard documentation techniques and abbreviations have been used. In addition, where abstracts exist, there follows a notation indicating where they are available. Since abstract publications are sometimes late in reporting (especially with foreign publications or during wartime conditions), the year of abstract availability is also included. The following symbols are used to indicate abstract publications: BSIR (*Bibliography of Scientific and Industrial Reports*, U. S. Department of Commerce), CA (*Chemical Abstracts*), CZ (*Chemisches Zentralblatt* and its predecessor *Chemisches Central-Blatt*), and EI (*Engineering Index*). Wherever possible, English titles have been used, except in the case of books or where a translation might have changed the meaning.

An attempt has been made to divide the various entries into fields of interest. However, this system is sometimes arbitrary and choosing the category was difficult for some items. It should be noted that duplicate or cross-listings are not used. Titles are often misleading and, in most cases, the abstract or article was read to determine either a definite solid propellant item or one of value to the field. Listings are alphabetical by author and each item has been numbered for reference purposes. While an attempt at completeness and accuracy was made, it is realized that some errors or deletions may have been inadvertently made. The author, therefore, welcomes comments about corrections or additions.

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[1] President. Mem. ARS.

1 Applications

In addition to the conventional rocket propulsive applications, a growing number of diverse applications has been noted. These began as early as 1871 (1.001) and now include power for autogyros and helicopters (1.008, 1.063), antispin rockets (1.013), valve operation (1.017), starters (1.018, 1.019, 1.053, 1.065, 1.086, 1.158), recoilless guns (1.010, 1.083), industrial applications (1.087, 1.088, 1.089, 1.133, 1.152, 1.155, 1.164), and a host of auxiliary devices (1.084, 1.110, 1.153, 1.165, 1.169).

Even the conventional application of solid propellants for rockets, missiles, and RATO seems to have been widely diversified: rainmaking (1.033); lifesaving and projecting (1.036, 1.122); astronautics (1.098, 1.127, 1.170), and research (1.041, 1.042, 1.045, 1.054, 1.057, 1.085, 1.106, 1.121, 1.136, 1.150). Hybrid applications have also appeared (1.077, 1.091, 1.159).

2 Ballistics, External

Theories of the external ballistics of solid propellant rockets have been published (2.010 2.015, 2.030, 2.031, 2.036, 2.040) and have been followed by general performance analyses (2.003, 2.004, 2.005, 2.015) and performance under actual conditions (2.007, 2.008, 2.011, 2.014, 2.033, 2.034).

3 Ballistics, Internal

Spurred on by applications of new, high energy propellants, the field of internal ballistics received early attention by such investigators as Bunsen in 1857 (3.060), Abel in 1869 (3.001), Berthelot in 1871 (3.035, 3.036), and Charbonnier in 1908 (3.069). Increasing attention to the combustion of propellants in guns lead to a study of burning in bombs. This work, uniting guns and rockets, was carried out by workers such as Muraour and co-workers (3.237–3.301) and by later workers such as Bourgoin (3.047), Bonner (3.045), and Crawford (3.090, 3.091). The relationship between bomb burning rates and rocket motors has also been investigated (3.317).

The importance of additives, catalysts, etc., on the burning of propellants has been recognized (3.012, 3.033, 3.056, 3.203, 3.257, 3.267, 3.276, 3.334) but not completely explained. Other phenomena noted: radiation (3.021, 3.034, 3.243, 3.318, 3.320), heat conduction and transfer (3.019, 3.030, 3.031, 3.032, 3.139), solid particles in the exhaust (3.050, 3.144, 3.229) and erosion of propellants (3.005, 3.152, 3.204, 3.252). Of increasing importance is the study of combustion instability (3.071, 3.072, 3.151, 3.174, 3.176, 3.383).

Recently, the long bifurcated paths between gun and rocket have drawn nearer with the use of leaking guns, recoilless weapons, low pressure guns, and rocket-assisted guns (3.083, 3.166, 3.194, 3.218, 3.308, 3.387). In addition, the two fields are now closely related by propellant combustion and the thermal properties of gases at high temperatures and pressures (3.022, 3.026, 3.038, 3.039, 3.052, 3.057, 3.061, 3.078, 3.121, 3.172, 3.185, 3.188, 3.213, 3.327, 3.355, 3.440). Indeed, the fields of liquid and solid propellants have drawn nearer with the newer hybrid systems (3.018, 3.054).

Various laws and mechanisms for propellant combustion have been suggested (3.047, 3.081, 3.092, 3.101, 3.122, 3.145, 3.155, 3.196, 3.247, 3.248, 3.315, 3.350, 3.351, 3.352, 3.364, 3.367). However, there are yet unsolved problems (3.140).

4 Explosives and Pyrotechnics

Because of the nature of many solid propellants and propellant materials, a brief section on explosives and pyrotechnics has been added. Handling explosive materials (4.003, 4.004, 4.012, 4.022, 4.030, 4.032, 4.034, 4.058) is certainly to be recognized. Certain phenomena in explosions (properties of gases etc.) are also of value in the propellant field (4.008, 4.029, 4.031, 4.057). Also, the general field of pyrotechnics is the forerunner of both the propellant and explosives field. Pyrotechnics are now used mainly in igniters and other components (4.004, 4.010, 4.024 4.038, 4.047, 4.050, 4.060).

5 General

The description of solid propellants in general rocket works (5.002, 5.036, 5.041, 5.042, 5.060, 5.062, 5.078, 5.081, 5.092, 5.107, 5.113, 5.114, 5.117, 5.119, 5.120) has been supplemented by a general book devoted solely to solid propellant rockets (5.132).

New solid propellant programs by various newcomers to the field have become more diverse (5.001, 5.006, 5.008, 5.010, 5.017, 5.024, 5.025, 5.088, 5.122). Recently, inert components of solid rocket motors have been described (5.033, 5.039, 5.048, 5.050, 5.065, 5.104, 5.124) New construction materials, such as plastics (5.030, 5.044) and ceramics (5.029, 5.054), have been employed.

6 Hazards and Safety

The hazards of propellant materials have been established: black powder (6.003, 6.054), homogeneous propellants (6.008, 6.037, 6.046, 6.050, 6.065, 6.071, 6.072), and other oxidants (6.001, 6.006, 6.015, 6.022, 6.033, 6.034). Sensitivity tests are also available (6.002, 6.042). Safety precautions and facilities practices have been established (6.005, 6.010, 6.012, 6.017, 6.024, 6.031, 6.039, 6.047, 6.059, 6.068).

7 History

Though much of the beginnings of solid propellants are as yet unknown, the early history has been well documented (7.007, 7.016, 7.017, 7.018, 7.023, 7.024, 7.025, 7.056). The intermediate stages of development have also been recorded (7.012, 7.015, 7.034, 7.038). Activities of the nineteenth

and twentieth centuries are also well covered. Items of historical interest are covered in several bibliographies (7.003, 7.022, 7.029, 7.030, 7.050, 7.052, 7.061), and a general review of the last twenty-five years of development is also available (7.055).

8 Ignition

Literature has started to appear on the ignition of solid propellants (8.006, 8.007, 8.018) and has resulted in several ignition theories (8.002, 8.015, 8.026). In addition, solid state ignition reactions have been studied (8.016, 8.022, 8.024) along with ignition by hot gases (8.004, 8.008).

9 Patents

In contrast to liquid propellants, most of the specifications for solid propellant compositions are to be found in a wealth of patent literature. By and far, the largest number are for smokeless powders. With the exception of gunpowders, there are only a few patents on composite propellants (9.026, 9.148, 9.187). It is interesting to note the artifices used in patent titles. For examples, the term solid rocket propellant is used sparingly (9.002, 9.036, 9.356) while such terms as propellant powder (9.005), propellant explosive (9.024), colloidal preparation (9.027), rocket powder (9.034), propulsive cartridge (9.040), readily combustible plastic (9.048), propellant charge (9.054), propellant fuel cartridge (9.062), and self-combustible charge (9.157) are used. The term self-accelerating projectile (9.112) has been used in place of the more simple word, rocket.

The use of solid propellants as gas generators has been the source of many patents: such novel uses as a starter for crude-oil engines (9.212), cartridge-operated valves (9.257), solid propellant helicopter (9.214) and a whole line of gas-producing compositions (9.302–9.309).

10 Physical and Analytical Chemistry

Most physical topics are concerned with the stability of propellants and propellant materials. The effect of stabilizers for homogeneous propellants has received considerable attention (10.001, 10.014, 10.016, 10.037, 10.063, 10.168). The use of pH as a parameter in the stability of double-base propellants has been studied (10.007, 10.114, 10.159). With homogeneous propellants such effects as ultraviolet radiation (10.022, 10.024), ozone (10.017), x-rays (10.140), molds (10.142), heat and humidity (10.147, 10.194) have been investigated. The physical and chemical stability of oxidants (10.027, 10.029, 10.088, 10.090, 10.093, 10.095, 10.107) has been studied and applied as parameters for composite propellant stability. In addition, such effects as catalysts (10.172) have received attention. With many composite oxidants, humidity control has been vital for producing satisfactory propellants (10.231).

Analytical techniques have been evolved to determine the various stabilizers (10.045, 10.109, 10.117, 10.136, 10.149) and solvents (10.060, 10.072) used in propellants. Evolved gas (10.021, 10.155) and nitrogen (10.097) have been used as parameters for the stability study of propellants. Newer analytical techniques are: chromatographic (10.006, 10.174), infrared (10.050, 10.184, 10.187), and polarographic (10.179).

11 Production and Processes

Previously much of the processing of solid propellants was covered by patents or simply kept in the trade secret domain. Recently, however, literature has appeared on double-base propellants (11.014, 11.020, 11.038, 11.040, 11.046, 11.047, 11.048) and composites (11.057, 11.058).

12 Propellants

A new era in solid propellants began with the transition from gunpowder to nitrocellulose (12.008, 12.024, 12.025, 12.097) and double-base propellants (12.005, 12.006, 12.065). There then followed a renewed interest in composites (12.089, 12.116, 12.125, 12.134, 12.135). There has been realization of the advantages of tailor-made propellants (12.147, 12.151). Also, the hybrid propellant systems (12.008, 12.095, 12.103) and the solid-fuel ramjet (12.122) appear attractive.

Data on modern propellants (12.009, 12.010) are sparse but there are some data on recent propellants of other nations (France: 12.113; Germany: 12.001, 12.015, 12.029, 12.041; Japan: 12.136, 12.139).

13 Propellant Materials

Many oxidants (13.001, 13.005, 13.017, 13.018, 13.019, 13.027), fuel-binders (13.004, 13.013, 13.015, 13.033), and related materials have received recent attention as propellant materials.

14 Testing

The ancient methods of testing have now been supplanted by modern instrumental techniques. Grains have been examined by x-ray and x-ray diffraction (14.108, 14.022), ultrasonics (14.016), and other nonvisual (14.029) methods. Environmental tests (14.048) have become common. Electronic methods (14.014, 14.015, 14.062) are used in measuring thrust and pressure during static tests. Microwaves have been used to determine flame temperatures (14.065), and high speed photography (14.001, 14.028, 14.077) has won a definite place in ballistics research. The general field of rocket test instrumentation (14.090) has been presented in detail. Dynamic test techniques have also been documented (14.054, 14.080, 14.093).

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9.020. Ball, A. M., Inhibitor Strips for Large-Grain Smokeless Powder, US 2,643,-611, June 30, 1953. CA 1953.

9.021. Banus, M. D., McSharry, J. J., Increasing the Burning Rate of Metal Powders, US 2,688,575, Sept. 7, 1954.

9.022. Barsky, G., Propellant Powders Containing Pelargonic Esters, US 2,396,074, Mar. 5, 1946. CA 1946.

9.023. Barsky, G., Flashless Smokeless Powder, US 2,439,281, Apr. 6, 1948. CA 1948.

9.024. Bart, C. D., Propellant Explosives, Brit. 590,270, July 14, 1947. CA 1950.

9.025. Bent, L. N., Smokeless Powder, US 2,112,257, Mar. 29, 1938. CA 1938.

9.026. **Bergen, J., Maire, G., Maurice, P.,** Preparation of Adhesive Plastic Masses, Fr. 1,040,557, Oct. 16, 1953, CZ 1954.

9.027. **Bergl, K.,** Colloidal Preparations, Ger. 541,468, Oct. 7, 1926. CA 1932.

9.028. **Berlin-Karlsruher Ind. Werk, AG,** Marking the Flight of Projectiles, Ger. 629,803, May 13, 1936. CA 1936.

9.029. **Boddicker, F. L.,** Smokeless Powder, US 2,186,516, Jan. 9, 1940. CA 1940.

9.030. **Bohlman, W. S., Bruce, D. S.,** Chromium Plated Rolls for Mills Used in the Production of Smokeless Powder, US 2,453,404, Nov. 9, 1948. CA 1949.

9.031. **Bohrn, W.,** Signal Rockets, Brit. 184,715, Sept. 19, 1921. CA 1923.

9.032. **Bonell, F. T. B., Juhlin, R.,** Preparation of Pressed Rocket Propellant, N.P. 76,261, Feb. 6, 1950. CZ 1951.

9.033. **Bonell, F. T. B.,** Preparation of Fast-Burning Propellants, Swed. 130,176, Nov. 21, 1950. CZ 1952.

9.034. **Bonell, F. T. B.,** Preparation of a Rocket Powder, Swed. 134,189, Jan. 8, 1952. CZ 1953.

9.035. **Bonell, F. T. B., Juhlin, R.,** Rocket Powder, US 2,622,277, Dec. 23, 1952. CA 1953.

9.036. **Bonell, F. T. B., Juhlin, R.,** Solid Rocket Propellant, E.PP. 655,586, Aug. 25, 1951. CZ 1954.

9.037. **Borland, C. R.,** Smokeless Powder, US 1,752,881, Apr. 1, 1930. CA 1930.

9.038. **Bowden, F. P., Blackwood, J. D.,** Sulfur-Free Black Powder, Brit. 715,827, Sept. 22, 1954. CA 1955.

9.039. **Bowden, F. P., Blackwood, J. D.,** Fast-burning Black Powder, Brit. 715,828, Sept. 22, 1954. CA 1955.

9.040. **Brandt, E. W.,** Propulsive Cartridge for Mortar Projectiles, US 2,052,546, Sept. 1, 1936. CA 1936.

9.041. **Broadbent, A. L., Woodbridge, R. G.,** Progressive Propellant Explosive, US 1,313,459, Aug. 19, 1919.

9.042. **Broadbent, A. L., Woodbridge, R. G.,** Propellant Powder, Can. 198,563, Mar. 23, 1920. CA 1920.

9.043. **Brooks, C. W.,** Black Powder, Can. 348,641, Mar. 5, 1935. CA 1935.

9.044. **Brooks, R. M.,** Nitration of Cellulose, US 2,678,310, Nov. 5, 1954. CA 1955.

9.045. **Brown, J. M.,** Smokeless Powder, Fr. 478,384, Dec. 8, 1915. CA 1916.

9.046. **Brown, J. M.,** Chlorate Explosive, US 1,218,976, Mar. 13, 1917.

9.047. **Brown, J. M.,** Smokeless Powder and Process of Making Same, US 1,222,065, Apr. 10, 1917. CA 1917.

9.048. **Brown, H. A.,** Readily Combustible Plastic, US 2,404,688, July 23, 1946. CA 1946.

9.049. **Bruce, D. S., Klein, E. L.,** Propellant Powder, US 2,499,295, Feb. 28, 1950. CA 1950.

9.050. **Burrows, L. A., Van Winter, C. B.,** Electric Squib, US 2,228,339, Jan. 14, 1941. CA 1941.

9.051. **Buxbaum, J.,** Black Gunpowder, US 1,293,236, Feb. 4, 1919. CA 1919.

9.052. **Cairns, R. W.,** Smokeless Powder, Can. 485,661, Aug. 12, 1952. CZ 1953.

9.053. **Cairns, R. W.,** Smokeless Powder, Can. 485,662, Aug. 12, 1952. CZ 1953.

9.054. **Cairns, R. W.,** Propellant Charge for Jet-Propelled Devices, US 2,643,184, June 23, 1953. CA 1953.

9.055. **Caldwell, W. A.,** Non-Porous Compressed Igniter, Swed. 130,593, Jan. 16, 1951. CZ 1953.

9.056. **Canadian Industries Ltd.,** Electrical Igniter, Oe.P. 169,341, Nov. 10, 1951. CZ 1952.

9.057. **Canadian Industries Ltd.,** Electrical Igniter, N.P. 80,297, June 3, 1952. CZ 1954.

9.058. **Cimorosi, A.,** Fireworks, US 2,174,202, Sept. 26, 1940. CA 1940.

9.059. **Claessen, C.,** Smokeless Powder, US 909,546, Jan. 12, 1909. CA 1909.

9.060. **Claessen, C.,** Smokeless Powder, Swiss 70,994, Nov. 16, 1915. CA 1916.

9.061. **Cocking, A. T.,** Smokeless Powder, US 903,958, Nov. 17, 1909. CA 1909.

9.062. **Cohan, A. M.,** Propellant Fuel Cartridge, US 2,429,548, Oct. 21, 1947. CA 1948.

9.063. **Crater, W. C.,** Smokeless Powder, US 2,024,128, Dec. 10, 1935. CA 1936.

9.064. **Crawford, B. L.,** Double-Base Smokeless Powder, US 2,440,327, Apr. 27, 1948. CA 1948.

9.065. **Crawford, B. L.,** Molded Propellant Powders, U. S. Pat. Appl. 742,463, *Off. gaz.*, vol. 624, 1949, p. 334. CA 1949.

9.066. **Curtis, C. H., Andre, G. G.,** Improvement in the Preparation of Smokeless Powder, Fr. 11,383. CZ 1892.

9.067. **Curtis, C. H., Andres, G. G.,** Improvement in the Preparation of Smokeless Powder, Brit. 19,068. CZ 1893.

9.068. **Davis, T. L.,** Coating Smokeless Powder, US 1,432,323, Oct. 17, 1922. CA 1923.

9.069. **Davis, T. L.,** Smokeless Powder, US 1,439,505, Dec. 19, 1922. CA 1923.

9.070. **Davis, T. L.,** Smokeless Powder, US 1,478,892, Dec. 25, 1923. CA 1924.

9.071. **Davis, W. R.,** Ammonium Nitrate Charges, US 2,509,812, May 30, 1950. CA 1950.

9.072. **Dehn, W. H.,** Propellant Explosive, US 1,819,458, Aug. 18, 1931. CA 1931.

9.073. **Delfino, P. P.,** Nitrocellulose Propellant Powder, US 2,096,451, Oct. 19, 1937. CA 1938.

9.074. **Demoix, P. E. J.,** Percussion Fuse for Rocket Projectiles, US 2,145,507, Jan. 31, 1939. CA 1939.

9.075. **Desetti, G. E., Drew, W. A., Mackey, B. H.,** Recovery of Solvents from Freshly Granulated Smokeless Powder, US 2,280,803, Apr. 28, 1942. CA 1942.

9.076. **Desetti, G. E., Drew, W. A., Mackey, B. H., Woodbridge, R. G.,** Apparatus for the Treatment of Smokeless Powder to Recover the Solvent Used, US 2,349,249, May 23, 1944. CA 1945.

9.077. **Dickerman, L. A.,** Smokeless Powder, US 2,120,324, June 14, 1938. CA 1938.

9.078. **Dovgelevick, N. I.,** Flameless Powder, Russ. 5039, Sept. 15, 1924. CA 1929.

9.079. **Drew, W. A., Mackey, B. H.,** Drying Smokeless Powder, US 2,281,560, May 5, 1942.

9.080. **E. I. du Pont,** Gunpowder Corning Mill, US 1,116,040, Nov. 3, 1914. CA 1915.

9.081. **E. I. du Pont,** Black Powder in Pellet Form, US 1,295,975, Mar. 4, 1919. CA 1919.

9.082. **E. I. du Pont,** Black Gunpowder, US 1,308,342, July 1, 1919. CA 1919.

9.083. **E. I. du Pont de Nemours & Co.,** Ballistic Powders, Fr. 698,308, July 2, 1930. CA 1931.

9.084. **E. I. du Pont de Nemours & Co.,** Ballistic Powders, Fr. 701,463, Aug. 30, 1930. CA 1931.

9.085. **E. I. du Pont de Nemours & Co.,** Propellant Powders, Brit. 360,950, Aug. 9, 1930. CA 1933.

9.086. **E. I. du Pont de Nemours & Co.,** Smokeless Powder, Ger. 567,878, June 26, 1930. CA 1933.

9.087. **E. I. du Pont de Nemours & Co.,** Ballistic Powders, Fr. 716,429, May 1, 1931. CA 1932.

9.088. **E. I. du Pont de Nemours & Co.,** Propellant Powders, Ger. 640,312, Jan. 5, 1937. CA 1937.

9.089. **E. I. du Pont de Nemours & Co.,** Treatment of Smokeless Powder, Brit. 550,325, Jan. 4, 1943. CA 1944.

9.090. **F. I. du Pont,** Recovery of Solvent from Smokeless Powder, US 1,294,066, Feb. 11, 1919. CA 1919.

9.091. **F. I. du Pont,** Smokeless Powder, US 1,308,343, July 1, 1919. CA 1919.

9.092. **F. I. du Pont,** Smokeless Powder, US 1,308,344, July 1, 1919. CA 1919.

9.093. **F. I. du Pont,** Smokeless Powder, US 1,308,346, July 1, 1919. CA 1919.

9.094. **F. I. du Pont,** Propellant Explosive, US 1,341,207, May 25, 1920. CA 1920.

9.095. **F. I. du Pont,** Smokeless Propellant Explosive, Brit. 298,543, Apr. 7, 1927. CA 1929.

9.096. **F. I. du Pont, E. du Pont,** Smokeless Explosive Powder, US 1,627,691, May 10, 1927. CA 1927.

9.097. **Dynamit, A G,** Igniter Material, Oe.P. 176,785, Nov, 25, 1953. CA 1954.

9.098. **Ellsworth, H. E.,** Black Gunpowder, US 1,770,482, July 15, 1930. CA 1930.

9.099. **Elsner, H.,** Weatherproof Electric Squib, DBP 803,645, Apr. 5, 1951. CZ 1952.

9.100. **Etat Francais,** Propellant Powder, Fr. 971,644, Jan. 19, 1951. CA 1952.

9.101. **I. G. Farben,** Esters of Sugar and Nitric Acid, Ger. 605,733, Nov. 17, 1934. CA 1935.

9.102. **Fenimano, C. P., Kuh, L. P., Frozen, J. H.,** Propellant, US 2,484,023, Oct. 11, 1949. CA 1950.

9.103. **Fidlar, J. B., Franklin, C. R.,** Propellant Explosive Powder, US 1,552,601, Sept. 8, 1925. CA 1925.

9.104. **Filbert, W. F.,** Igniter Mixture, US 2,511,669, June 13, 1950. CZ 1952.

9.105. **Fletcher, H., Tschudin, E.,** Nitrocellulose, Ger. 555,765, Mar. 26, 1929. CA 1932.

9.106. **Foster, G. H., Williams, E. F.,** Crystalline Material, US 2,445,478, July 20, 1948. CA 1948.

9.107. **Franklin, C. R.,** Propellant Explosive Powder, US 1,547,868, July 28, 1925. CA 1925.

9.108. **Franklin, C. R.,** Propellant Explosive, US 1,564,549, Dec. 8, 1925. CA 1926.

9.109. **Franklin, C. R.,** Nitrocellulose Propellant Powder, US 1,582,256, Apr. 27, 1926. CA 1926.

9.110. **Frejacques, J. L. M.,** Preparation of Potassium Chlorate Crystals, US 2,595,238, May 6, 1952. CZ 1953.

9.111. **Gaens, F.,** New Propellant, Ger. 37,631, Oct. 14, 1885. CZ 1887.

9.112. **Ganahl, C. D.,** Self-Accelerating Projectiles, US 2,408,252, Sept. 24, 1946. CA 1947.

9.113. **Garcia, A. B.,** Stabilization of Nitrocellulose, Span. 210,044, Oct. 19, 1953. CA 1955.

9.114. **Gehrig, W. F.,** Pyrotechnical Composition, US 1,830,392, Nov. 3, 1931.

9.115. **Geitmann, J. T.,** Device for Producing Radially Unsymmetrical Cords or Tubes of Smokeless Powder, Ger. 715,246, Nov. 20, 1941. CA 1944.

9.116. **Goehmann, M. H.,** Recoilless Mortar, US 2,406,537, Aug. 27, 1946.

9.117. **Goodyear, E. S.,** Smokeless Powder, US 2,076,772, Apr. 13, 1937. CA 1937.

9.118. **Goodyear, E. S.,** Progressive-Burning Smokeless Powder, US 2,147,698, Feb. 21, 1939. CA 1939.

9.119. **Goodyear, E. S.,** Smokeless Powder, US 2,179,330, Nov. 7, 1939. CA 1940.

9.120. **Goodyear, E. S.,** Propellant Powder, US 2,228,309, Jan. 14, 1941. CA 1941.

9.121. **Goodyear, E. S.,** Propellant Powder, Brit. 544,705, Apr. 24, 1942.

9.122. **Goodyear, E. S.,** Propellant Smokeless Powder, US 2,346,125, Apr. 11, 1944, CA 1944.

9.123. **Gordon, S., Whitworth, E.,** Flashless Propellant Powders, Brit. 616,898, Jan 28, 1949. CA 1949.

9.124. **Gordon, S.,** Granular Smokeless Propellant Powders, Brit. 626,955, July 25, 1949. CA 1950.

9.125. Gordon, S., Whitworth, E., Smokeless Powder, Fr. 953,268, Dec. 2, 1949. CA 1950.

9.126. Gordon, S., Whitworth, E., Smokeless Propellant, Swed. 130,177, Nov. 2, 1950. CZ 1952.

9.127. Gordon, S., Whitworth, E., Granular Smokeless Propellant Powder, US 2,575,-871, Nov. 20, 1951. CA 1952.

9.128. Goudy, R. C., Smoke Making Composition for Signal Rockets, US 1,318,074, Oct. 7, 1919. CA 1920.

9.129. Goudy, R. C., Smoke Making Composition for Signal Rockets, US 1,319,329, Oct. 21, 1919. CA 1920.

9.130. Gray, J. P., Slow Burning Black Powder Explosive, US 1,438,759, Dec. 12, 1922. CA 1923.

9.131. Gunther, U., Primer for Igniting, US 2,687,667, Aug. 31, 1954.

9.132. Habbel, H., Weatherproof Electric Squib, DBP 813,137, Sept. 6, 1951. CZ 1952.

9.133. Habbel, H., Weatherproof Electrical Squib, Ger. 889,575, Sept. 1953. CZ 1955.

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9.135. Hale, G. C., Olsen, F., Propellant Explosive, US 1,547,808, July 28, 1925. CA 1925.

9.136. Hale, G. C., Propellant Powder, US 1,899,147, Feb. 28, 1933. CA 1933.

9.137. Hale, G. C., Cameron, D. R., Nitrocellulose Propellant Powder, US 2,035,-471, Mar. 31, 1936. CA 1936.

9.138. Hale, G. C., Cameron, D. C., Propellant Nitrocellulose Explosive, US 2,026,-531, Jan. 7, 1936. CA 1936.

9.139. Hale, G. C., Smokeless Powder, US 2,440,267, Apr. 27, 1948. CA 1948.

9.140. Hale, G. C., Audrieth, L. F., Propellant Powders, US 2,480,852, Sept. 6, 1949. CA 1950.

9.141. Hanley, E. J., Ignition Composition, US 2,476,370, July 19, 1949. CA 1949.

9.142. Hart, D., Eppig, H. J., Powers, W. J., Self-Hardening Pyrotechnic Composition, US 2,700,603, Jan. 25, 1955. CA 1955.

9.143. Hassel, O., Hexamethylenetetramine perchlorates, Nor. 57,831, Apr. 12, 1937. CA 1937.

9.144. Hawkesworth, A. S., Nitrocellulose Smokeless Powder, US 1,713,505, May 14, 1929. CA 1929.

9.145. Hechler, G., Continuous Extraction of Water from Nitrocellulose by Means of Alcohol, D.BP 820,107, Aug. 11, 1951. CZ 1953.

9.146. Hengst, C. F., Smoke and Flameless Propellant, Ger. 48,265. CZ 1890.

9.147. Henning, C. I. B., Propellant Explosives, US 1,357,865, Nov. 2, 1920. CA 1921.

9.148. Henning, C. I. B., Propellant Smokeless-Powder Charge, US 1,393,623, Oct. 11, 1921.

9.149. Hercules Powder Co., Drying Smokeless Powder, Fr. 820–059–60, Nov. 2, 1937. CA 1938.

9.150, 9.151. Hercules Powder Co., Smokeless Powder, Brit. 508,914, 528,035; July 7, 1939, Oct. 22, 1940. CA 1940, CA 1941.

9.152. Hercules Powder Co., Progressive-Burning Smokeless Powder, Brit. 536,096, May 2, 1941. CA 1942.

9.153. Hercules Powder Co., Smokeless Powder, Brit. 621,685, Apr. 14, 1949. CA 1954.

9.154. Hickman, C. N., Propellant Having an Opacifier for Preventing Self-Ignition by Radiation, US 2,574,479, Nov. 13, 1951. CA 1952.

9.155. Hickman, C. N., Blending Powder Grains, US 2,437,694, Mar. 16, 1948. CA 1948.

9.156. Höhne, H., Pegels, K., Riedel, M., Elimination of Water From an Aqueous Solution of Nitrocellulose by Means of Centrifuge, DBP 823,119, Nov. 29, 1951. CZ 1953.

9.157. Holm, J. M., Imperial Chemical Ind., Self-Combustible Charges for Generating Gases under Pressure, Brit. 477,956, Jan. 6, 1938. CA 1938.

9.158. Holm, J. M., Imperial Chemical Ind., Gas Producing Cartridges for Mechanical Operations, Brit. 483,994, Apr. 22, 1938. CA 1938.

9.159. Holm, J. M., Combustion Cartridge for the Slow Generation of Gas Pressure, US 2,165,263, July 11, 1938. CA 1939.

9.160. Holmes, H. H., Smokeless Powder with Stearate Glaze, US 2,385,135, Sept. 18, 1945. CA 1946.

9.161. Horst, A. G., Slow Burning Black Powder, Russ. 41,889, Feb. 28, 1935. CA 1936.

9.162. Hough, A., Smokeless Powder Propellant, US 1,916,763, July 4, 1933. CA 1933.

9.163. Hutchinson, A. C., Solid Gas Producing Mixture, Fr. 970,531, Jan. 5, 1951. CZ 1952.

9.164. Hutchinson, A. C., Solid Gas Generating Units, US 2,710,793, Jan. 14, 1955. CA 1955.

9.165. Ingraham, W. T., Smokeless Powder, US 1,710,024, Apr. 23, 1929. CA 1929.

9.166. Ingraham, W. T., Smokeless Powder, US 1,731, 378, Oct. 5, 1929. CA 1930.

9.167. Imperial Chemical Ind. of Australia, Gas Producing Non-Detonating Charges, Australia 100,512, Mar. 25, 1937. CA 1937.

9.168. Imperial Chemical Industries, Solid Propellant, Denmark 73,162, Oct. 10, 1947. CZ 1954.

9.169. Imperial Chemical Industries. Nitropolystyrol, Swiss 275,797, Sept. 1, 1951. CZ 1952.

9.170 Imperial Chemical Industries, Igniter, Swiss 271,938, Feb. 16, 1951. CZ 1952.

9.171. Imperial Chemical Industries, Solid Propellant, N.P. 78,328, Mar. 27, 1951. CZ 1954.

9.172. Imperial Chemical Industries, Ignition Cable, N.P. 78,417, Apr. 9, 1951. CZ 1953.

9.173. Imperial Chemical Industries, Electrical Igniter, Oe.P 177,103, Dec. 28, 1953. CZ 1954.

9.174. Imperial Chemical Industries, Electrical Igniter, Oe.P 178, 577, May 25, 1954. CZ 1955.

9.175. Ipartelepek, N., Nitrocellulose Powder, Hung. 118,216, June 15, 1938. CA 1938.

9.176. Jackson, H., Miles, F. D., Nitroguanidine, Brit., 584,107, Jan. 27, 1947. CA 1947.

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9.178. Johnson, J. R., Blomquist, A. T., Nitrate Esters in Powders, US 2,653,165, July 7, 1954. CZ 1955.

9.179. Jones, E., Forlin, Zundschnur, F. P. 990,030, Sept. 17, 1951. CZ 1953.

9.180. Jones, G., Ignitable Composition for Matches, US 1,201,191, Oct. 10, 1916. CA 1916.

9.181. Keck, C. H., Propellant Explosive Containing Nitrocellulose and Nitrated Castor Oil, US 1,651,459, Dec. 6, 1927. CA 1928.

9.182. Keck, C. H., Nitrocellulose Propellant Explosive, US 1,784,189, Dec. 9, 1930. CA 1931.

9.183. Kenney, J. F., Ammunition Priming Mixture, US 2,689,788, Sept. 21, 1954.

9.184. Kniazykowski, E. B., Partyka, C., Smokeless Powder, Fr. 643,857, Nov. 15, 1927. CA 1929.

9.185. Knutson-Hall, F. K., Repeatedly Ignitable Pyrotechnic, US 2,568,285, Sept. 18, 1951. CA 1952.

9.186. Lamarre, A., Improvement in Pyrotechnic Signals, US 144,030, Oct. 28, 1873.

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9.188. Leuschel, R. F., Process for Production of Propellant, Ger. 105,877, 23/11, 1898. CZ 1900.

9.189. Lewers, W. W., Menges, W. A., Pittman, E. C., Smokeless Powder Treatment, US 2,302,558, Nov. 17, 1942. CA 1943.

9.190. Liedbeck, A., Process for Making Nitrocellulose, Ger. 96,109. CZ 1898.

9.191. Lindsley, M. F., Smokeless Powder, US 2,031,659, Feb. 25, 1936.

9.192. Lindsley, M. F., Smokeless Powder, US 2,247,392, July 1, 1941. CA 1941.

9.193. Linsten, D. C., Coating for Propellant Powders, US 2,432,578, Dec. 16, 1947. CA 1948.

9.194. Livingston, R., Hutchinson, A. C., Propellant Gas Producer Unit, Especially for Rocket Propulsion, Ger. 816,626, Oct. 11, 1951. CA 1953.

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9.197. Mackey, B. H., Smokeless Powder, US 2,342,304, Feb. 22, 1944. CA 1944.

9.198. Mackey, B. H., Milus, P. R., Smokeless Powder, US 2,349,048, May 16, 1944. CA 1945.

9.199. Maisner, H., Process for Gelling Liquid Nitroparaffins, US 2,690,964, Oct. 5, 1954.

9.200. Mangram, S. J., Blissel, J. J., Mixture for Igniter, US 2,640,770, June 2, 1953. CZ 1954.

9.201. Marsh, H. N., Smokeless Powder, US 2,131,352, Sept. 27, 1938. CA 1938.

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9.203. Mauis, G. H., Stieg, F. B., Nitrocellulose Smokeless Powders, US 1,498,053, June 17, 1924. CA 1924.

9.204. Maxin, H. S., Propellant, Ger. 37,430, June 23, 1885. CZ 1887.

9.205. McGabie, F. H., Perforated Powder Grains, Ger. 95,792. CZ 1898.

9.206. McIntyre, W. A., Black Powder, Can. 326,148, Sept. 20, 1932. CA 1932.

9.207. McIntyre, W. A., Black Powder Having a Slow Combustion Rate, US 1,882,853, Oct. 18, 1932. CA 1933.

9.208. McNutt, J. D., Propellant Explosive, US 1,858,373, May 17, 1932. CA 1932.

9.209. Melick, J. M., Induction Firing Means for Rockets, US 2,690,124, Sept. 28, 1954.

9.210. Menke, H., Igniter, Ger. 876,823, May 18, 1953. CZ 1955.

9.211. Mitchell, R. B., Removing Diphenylamine From Smokeless Powders, US 1,876,-726, Sept. 13, 1932. CA 1933.

9.212. Möhrle, A., Starting Cartridge Suitable for Starting Crude-Oil Engines, US 1,877,936, Sept. 20, 1932. CA 1933.

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9.216. Nathan, F. L., Rintoul, W., Gelatinized Propellant Explosive, US 1,302,202, Apr. 29, 1919. CA 1919.

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9.219. Nitrochemie Industrieanlagen, Appliance for the Production of Tubular Pow-

der, Ger. 712,548, Oct. 27, 1941. BSIR 1947.

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9.222. **Nitrokemia Ipartelepek, R. T.,** Manufacture of Gunpowder without Solvents, Hung. 128,458, Aug. 17, 1942. CA 1948.

9.223. **Nitrokemia Ipartelepek, R. T.,** Gunpowders of High Combustion Velocities, Hung. 131,592, Apr. 15, 1943. CA 1948.

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Book Reviews

Ali Bulent Cambel, Northwestern University, *Associate Editor*

Stresses in Aircraft and Shell Structures, by Paul Kuhn, McGraw-Hill Book Co., New York, 1956, xx + 435 pp. $14.

Reviewed by SENG-LIP LEE
Northwestern University

This book is a well correlated compilation of some of the research work in the field of aircraft and stiffened-shell structures performed in the NACA Structures Research Division.

The content is divided into two parts. The methods of analysis given in Part 1 are evaluated in the light of experimental evidence presented in Part 2. It is assumed that the reader has a working knowledge of statically indeterminate structures and ordinary differential equations.

Chapter 1 discusses the complexity of aircraft structures and the idealization of such structures by simplifying assumptions in order to render an analysis at all possible. The number and nature of these assumptions determine the accuracy and the complexity of the analysis. The tedious labor involved in the solution of an idealized structure containing a large number of structural elements can further be reduced by the use of a simplified or substitute structure in which certain common structural elements are replaced, for the purpose of analysis, by a single element. This results in further loss of details of the stress distribution but a simpler solution is obtained yielding reasonable approximations valuable for preliminary design purposes.

"Elementary theories" dealing with the shear flow and flexural stresses in thin-walled sections, with and without stringers, are given in Chapter 2. Both statically determinate and indeterminate sections as well as their shear centers are discussed. The corrections for taper effects in beams of varying section and simple cases of conical shells are also considered.

The "elementary theories" are based on the Navier-Bernoulli assumption that plane sections remain plane in bending, or the Bredt theory of the thin-walled torsion tube and certain restrictive assumptions. They may become significantly inaccurate in cases such as the bending of thin wings under transverse loading, whereby the presence of shear strains causes plane sections to warp out of their planes. In most shell structures, this warping tendency is restrained due to variation of the cross sections or the load along the span. The restraints change the stress distribution and the stiffnesses of the shell in bending and torsion, causing significant deviations from the "elementary theories." These deviations are analyzed in subsequent chapters.

A beam with thin shear web and transverse stiffness does not fail when the web buckles between the stiffeners; it merely forms diagonal folds which function as tension diagonals with the stiffeners acting as compression struts and the flanges acting as chord members. For very thin web, buckling takes place at a small fraction of the ultimate load and the state of stress approaches the limiting case of pure diagonal tension demonstrated theoretically by Wagner. In present-day aircraft practice, most practical web systems remain, even near failure, in a state of stress intermediate between the state of pure diagonal tension and the state of stress which exists before the web buckles.

This state of stress, termed incomplete diagonal tension, is treated in Chapter 3 where both plane and curved web systems are discussed. Empirical formulas and charts for estimating the strength of web, uprights and rivets are given and design procedure recommended. The structural efficiencies of several plane web systems are compared in diagrams useful as an aid for choosing the most efficient design. The interaction between torsion and axial compression in curved web systems is also investigated.

Consider a panel consisting of longitudinal edge flanges and intermediate stringers connected by sheet bays stiffened by transverse ribs. Under the action of longitudinal loads applied at the flanges, the stresses in the flanges and the stringers would be equal if the shear stiffness of the panel were infinite. The shear stiffness of the sheet, however, is finite. Under the action of the shear force which transmits a portion of the loads from the flanges to the stringers, the sheet deforms resulting in nonuniform strains, hence stresses, in the stringers. This phenomenon is commonly termed shear-lag effect.

Chapter 4 presents the analytical solutions of single-stringer panels of constant cross section with infinite as well as finite transverse stiffnesses. Numerical methods for treating panels with variable cross section are derived by the principle of consistent deformations in the form of recurrence formula in which the correction couples at the stations between bays, representing the shear-lag effect, are the redundant quantities. These solutions are employed in the analysis of box beams with single-stringer covers. Simplified analysis of multistringer panels by means of substitute single-stringer method is discussed and the procedure is extended to the analysis of fuselage and wing sections.

Chapter 5 investigates the interaction between torsion and bending moment in a two-spar wing structure which consists of two cantilever spars with connecting ribs. The covering skin is assumed to be ineffective, applicable in cases where the wings are covered with canvas or light plywood or where the metal covers are rendered structurally ineffective by large cutouts. Two-spar structure with a single rib at the tip is first treated and the influence of torsional and bending stiffnesses of the rib analyzed. Cases with multiple ribs are solved by means of the Karman-Friedrichs equations, a set of simultaneous linear equations in which the unknowns are the relief moments acting on the spars by the action of the ribs. The solution is extended to sweepback and nonparallel spars.

The torsion bending of four-flange shells under torsion is treated in Chapter 6. Doubly symmetrical rectangular section with finite bulkhead spacing is analyzed by means of a recurrence formula similar to that used in Chapter 4. The shell is divided into bays between the bulkheads and the deformation of each bay, first assumed to be free from warping restraint, is evaluated. To preserve continuity between adjacent bays, bicouples of indeterminate magnitudes representing the warping restraints between bays are applied at the faces of the cuts and the deformation in each bay thus produced determined. Equating the warps of the inboard and outboard ends of two consecutive bays meeting at each bulkhead leads to a set of simultaneous linear equations from which the magnitudes of the bicouples are calculated.

Shells with infinitesimal bulkhead spacing are also considered and the extension of the above-mentioned procedure to nonsymmetrical and curved-walled sections as well as shells tapered in width and depth is demonstrated. The application of the torsion bending theory to the analysis of sesquispar wing is briefly discussed.

The procedure just discussed is utilized in the analysis of multistringer shells under torsion taken up in Chapter 7. Three methods based on different simplifying assumptions concerning the distribution of constraint stresses over the cross section are presented. The wall-bending theory, applicable to rectangular section, assumes that the flexural stress distribution in the covers and the webs follows the linear law. The torsion-warping theory assumes that the longitudinal constraining stresses acting on a cross section are proportional to the warping that would exist if the tube were free of constraint. The linear-warping theory also assumes that these stresses are proportional to the warping displacement and that a plane section warps into a hyperbolic paraboloid surface. Numerical examples are given to illustrate the application of the procedure.

The method used in dealing with shear-lag problems is extended in Chapter 8 to analyze some fundamental cases of cutout problems in plane panels. Doubly symmetrical panels with rectangular cutouts are solved by means of the substitute stringer method; axial, shear and moment loadings are considered. In the three-stringer method, the coaming stringer which forms the longitudinal edge of the cutout is retained in the analysis. The stringers between the coaming stringer and edge flange, including the flange, are replaced by a substitute stringer while the stringers between the coaming stringer and the longitudinal center line are replaced by another.

Further simplification is attained in the two-stringer method by combining the former substitute stringer and coaming

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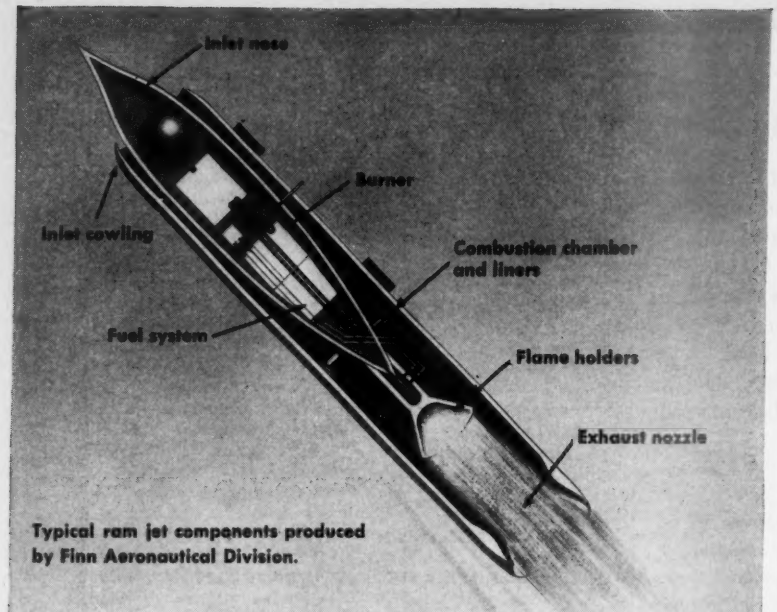

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stringer into a single substitute coaming stringer. The net section at the cutout is analyzed with the aid of empirical formula. Analysis of box beams with cutouts under moment and torque loads by means of the above procedure and the results of the last two chapters is discussed with particular reference to the modification of the flexibility coefficients in the recurrence formula to take the presence of cutouts into account.

Chapter 9 presents methods for estimating the stresses around rectangular cut outs in circular cylinders. Empirical methods are employed to analyze the net section and, for the gross section, adaptations of the semiempirical methods of the previous chapter are used. Both four-longeron cylinders and cylinders without longerons are discussed.

It can hardly be overemphasized that no method of analysis of a complex structure can possibly be exact. Many of the methods presented in this book are empirical or semiempirical in nature, while the others are based upon simplifying assumptions. Experimental evidence of their accuracy is presented in Part 2 where extensive comparisons between calculated and experimental results are given in the forms of graphs and tables. The range within which the methods may be applied with the expectation of reasonable accuracy may well be defined by the range of the tests.

This book should be valuable to practicing engineers in the field of aircraft structures as well as graduate students interested in this field.

Gaseous Nebulae, by L. H. Aller, Vol. 3 of the International Astrophysics Series, John Wiley & Sons, New York, 1956, 322 pp. $11.

S. F. Singer
University of Maryland

This is the third volume in a series of which two have already appeared, "The Aurora" by L. Harang and "Comets and Meteor Streams" by J. G. Porter. It is likely that the present volume will not be of such wide interest as were the first two books which deal more closely with the phenomena of our own atmosphere. The present volume is addressed to a rather specialized audience and one needs a background in astronomy and astrophysics, especially spectroscopy, in order to absorb very much of the text.

The information is presented in well organized and attractive fashion. The author, who has also written some successful text books on astrophysics, writes here about a field in which he himself has done a great deal of research. Nebulae are diffuse telescopic objects which cannot be resolved into stars. They include both clouds of particles and gas, and even whole stellar systems. But the author deals with clouds of particles and gas in our own galaxy, i.e., galactic nebulae of which the Orion nebula is probably the best known. Aller describes briefly methods of optical observation, both telescopic and photoelectric. Radiotelescopic observations are not yet well enough developed to give information on galactic nebulae. The author then describes the distances, di-

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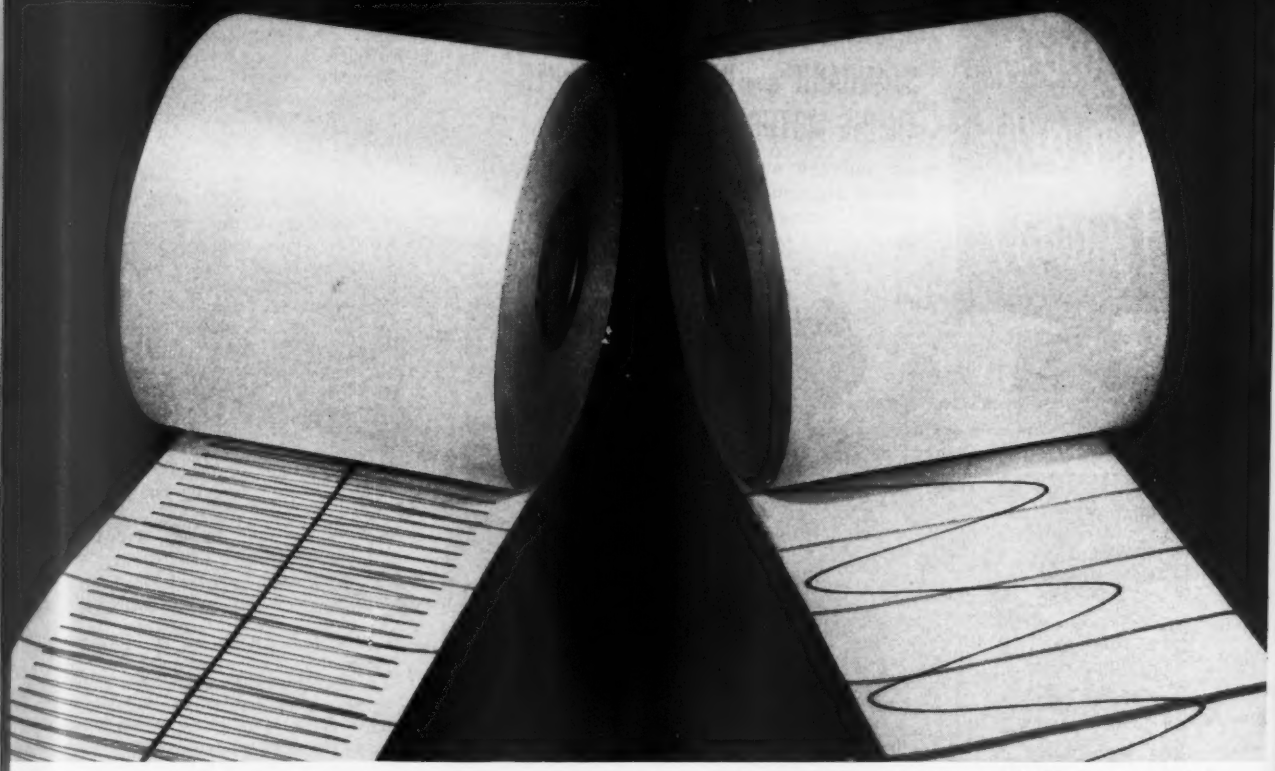

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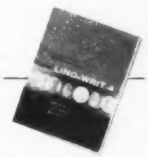

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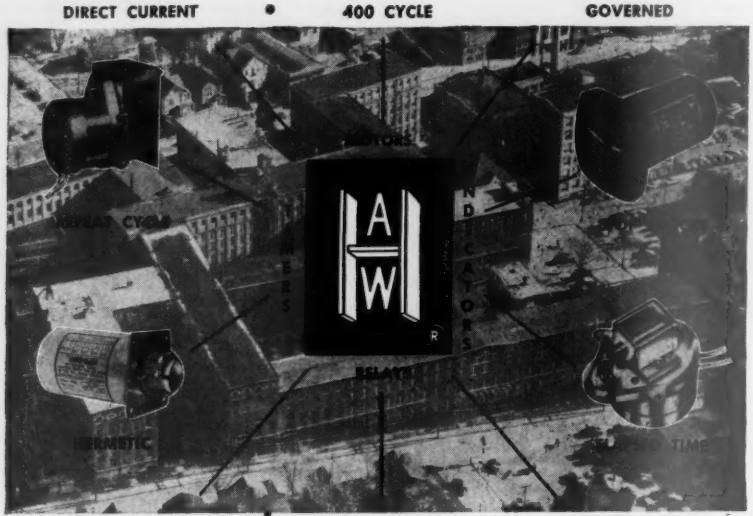

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mensions and spectra of the gaseous nebulae, as well as the physical processes within the nebulae which give rise to the luminosities and to the details of the spectra. He discusses forbidden lines, i.e., spectra lines which are normally not observed in laboratory experiments because there the pressures are too high. The nebulae, as might be guessed, derive all their energy from stars which are placed near or within them. The character of these stars is discussed in a separate chapter. Next the author takes up the structure and internal motions and finally, in more detail, the characteristics of diffuse galactic nebulae, which are really only denser regions of the interstellar medium. Here magneto-hydrodynamics becomes of some importance and the author reviews briefly the work on galactic magnetic fields, on turbulence in interstellar space and on its relation to the production of cosmic rays and of radio noise.

The book is a must in the library of every serious student of astrophysics.

Problems of Vision in Flight at High Altitude, by Thomas C. D. Whiteside, Butterworths Scientific Publications, London, England, 1957, 162 pp. $5.

Reviewed by WILLIAM F. ASHE
Ohio State University

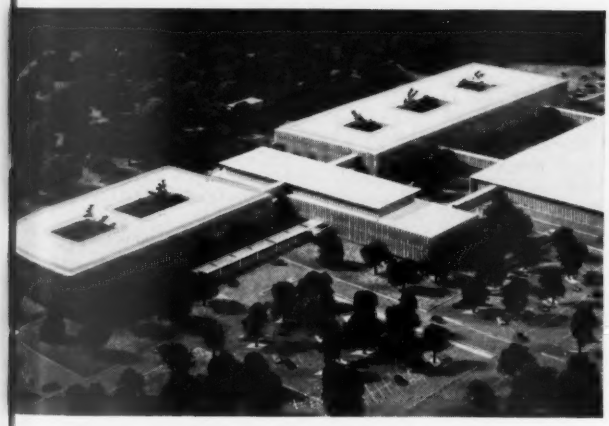
Thomas Whiteside has made a study of vision at 40,000 ft at a 90 deg angle to the sun through plexiglass canopy of jet fighter planes. The subjective problems are haze, glare, inability to see instruments well in cockpit and sudden recognition of an object in the sky which should have been seen sooner. The work was carefully done, is precisely described and limitations of instrumentation probably do not negate most of the conclusions drawn.

Difficulties in seeing instruments are due largely to the fact that most of the light comes from reflection from the cloud bank below and casts a darker shadow on instruments. It is the reverse of the terrestrial situation. Reversed light distribution is also largely responsible for the glare problem. Failure to see objects outside the plane is due to the fact that in an empty visual field the eye cannot remain accommodated for infinity, but becomes quite myopic and focuses well short of infinity. He attributes the persistent haze seen to be due at least in part to slight anoxia; this effect is demonstrable after a considerable period of exposure when the partial pressure of oxygen available is lowered only very slightly (equivalent cabin altitudes as low as 4000 ft). He did not prove the phenomenon would disappear if cabin altitude and PO_2 were maintained at sea level.

Older pilots will do a better visual job looking out of plane than younger subjects and about equally well for cockpit visual tasks providing their near vision is properly corrected. Dr. Whiteside's book substantiates the growing conviction that the day of "see and be seen" in flight is past. Traffic control and electronic object finding outside the plane are a must.

Human factors research teams and instrument panel design and test engineers need to read this book.

Dr. Dorothy M. Simon, Technical Assistant to the President of Avco Research and Advanced Development Division, addresses a colloquium of the Advisory Group for Aeronautical Research and Development (NATO) in Cambridge, England.



The Science Center for Avco's Research and Advanced Development Division, as seen in the architects' scale model, is now under construction in Wilmington, Massachusetts, a suburb 15 miles northwest of Boston's cultural hub. Scheduled for completion in early 1958, this ultramodern laboratory will house the scientific and technical staffs of the division.

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Dorothy M. Simon

Dr. Dorothy M. Simon
Technical Assistant to the President

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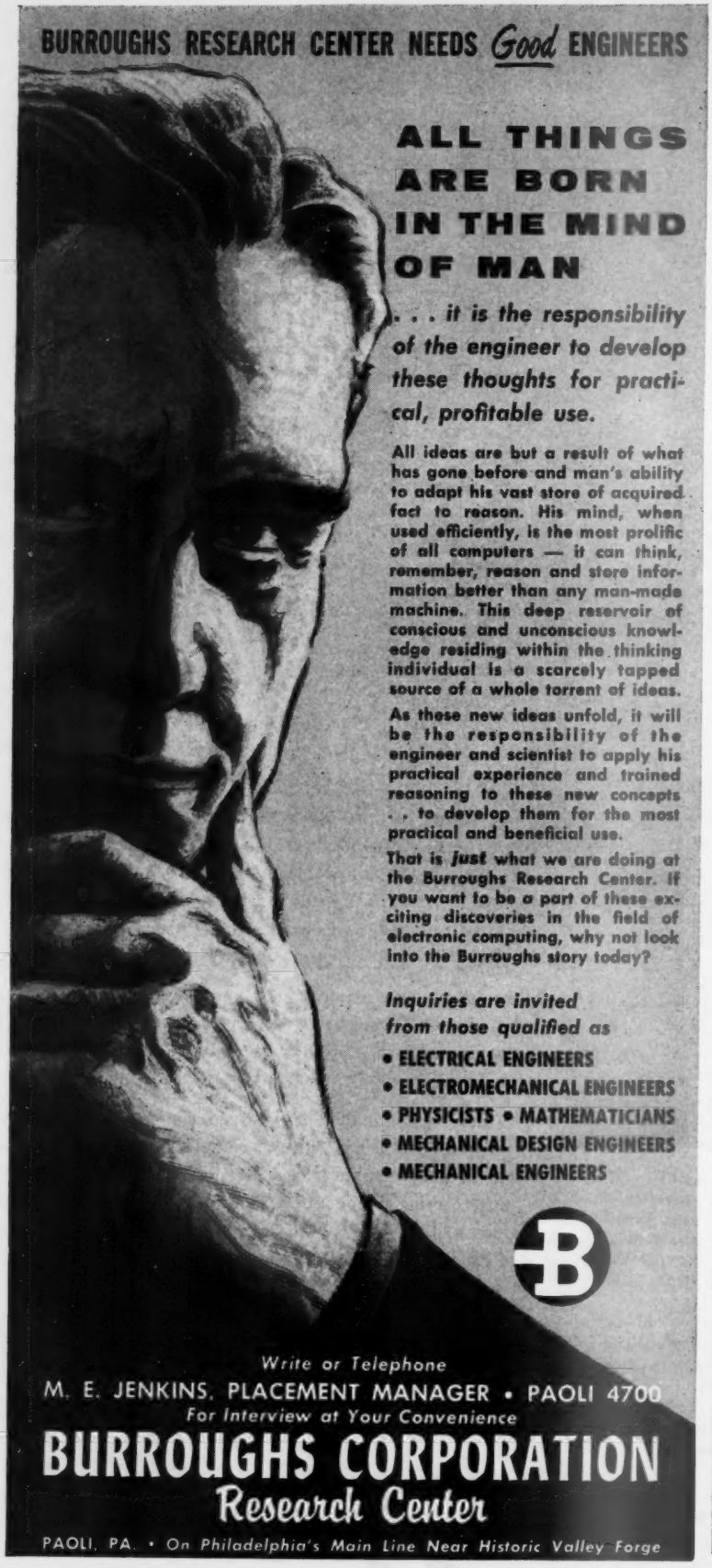
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Principles of Turbomachinery, by D. G. Shepherd, The Macmillan Company, New York, 1956, 463 pp. $10.

Reviewed by Robert B. Banks
Northwestern University

For some time there has been a definite need for a new textbook on the subject of turbomachinery. During the past decade there have been several texts on this subject; however, for the most part these books have been concerned with one or another specific aspects of the broad subject of turbomachinery. A new text on the subject might well be one which combines the pertinent fundamentals of fluid mechanics and thermodynamics with the considerable amount of design and performance knowledge which has now been acquired. The book which D. G. Shepherd has written appears to be such a needed text.

The author has divided his book into two general categories. The first half is related to fundamentals, the latter half to the specific features of the various types of turbomachinery. Following an introductory first chapter, the author deals at considerable length with the subject of dimensional analysis. Such early emphasis on this subject is judicious as it is bound to create in the student's mind the considerable role which such analysis plays in the dimensionless representation of machinery performance. Subsequently, the topic of energy transfer between a fluid and a rotor is developed. Such presentation is made by equating torque to change in angular momentum, which leads directly to the Euler equation. The latter is then applied to various basic types of turbomachinery for determination of energy developed or extracted by the machine.

The next two chapters are devoted to thermodynamics and to fluid flow. Needless to say, treatment of these subjects is necessarily brief and is restricted to the features of these subjects related to turbomachines.

The second portion of the book deals with the general types of machines. A logical category of types has been selected: Radial-flow pumps, radial-flow turbines, axial-flow pumps and axial-flow turbines. The first chapter of this latter portion is concerned with centrifugal pumps and compressors, and includes topics on the fundamentals of design of the inlet section, the impeller channel and the discharge casing of the basic radial-flow pump and compressor. In addition, there is a brief treatment on series and parallel operation. Following this, a chapter is devoted to radial-flow turbines, and includes discussions on the Ljungstrom, Francis and Banki types. The subject of axial-flow turbines is considered next; it incorporates the more important design fundamentals of gas and steam turbines as well as the propeller, Kaplan and Pelton wheel types. Finally, a chapter deals with axial-flow compressors and pumps. This is devoted to a discussion of the various methods of analysis and design of the axial-flow machine, including the isolated flat plate method, the lift-and-drag method and the direct cascade design method. Effects of Reynolds and Mach numbers are presented and the phenomena of stalling and

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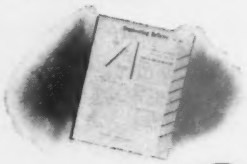

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surging are considered. Two additional chapters are included, one on the performance of turbines, the other on the performance of compressors and pumps. These chapters present comparisons of types of turbomachines and include considerable information on efficiencies and other performance characteristics.

The author has included problems at the end of the four chapters devoted to the fundamentals. In addition, at the conclusion of each chapter, he has presented a very comprehensive bibliography. Perhaps it would have been desirable to have included a chapter on fluid couplings and torque converters to round out his specific aim for a text on "dynamic" machinery. The book will be easily understood by one who has a knowledge of elementary fluid mechanics and thermodynamics. It would be appropriate for a senior or first-year graduate course in turbomachinery; the graduate engineer seeking to study the subject will be able to follow it without difficulty. The author has presented a comprehensive well-organized and outstanding book on the subject of turbomachinery.

Jets, Wakes and Cavities, by G. Birkhoff and E. H. Zarantonello, Academic Press, New York, 1957, 354 + xii pp. $10.

Reviewed by SHIH-I PAI
University of Maryland

One of the most interesting classes of the fluid flow problem is that of the free boundary problem which includes the problems of jets, wakes and cavities. Like other fluid flow problems, it is not possible to discuss this flow problem in a most general manner. One has to make proper assumptions to simplify the problem so that a special class may be analyzed at one time. One such assumption is that the flow may be considered as inviscid and incompressible so that Euler's differential equations may be applied. Such a special class of free boundary flow problems has been extensively investigated by many mathematicians since the first analysis by Helmholtz in 1868. Fruitful results have been obtained. Engineers are also interested in this problem because it offers explanations of some phenomena in the cavities produced by an obstacle in the liquid or liquid jets in air. The major portion of the book has been devoted to these potential flow problems of Helmholtz and their generalizations. It has been well written and may be considered as the first complete and systematic treatment of such problems.

There are many other interesting fluid flow problems of free boundary flow, such as the jet mixing problems, base pressure problem and atomization of jet. The authors discuss very briefly these problems from the mathematicians point of view. The material of the other free boundary flow problems discussed in the book is rather insufficient from the engineering point of view. For instance, a single small section is given about base pressure problem which should be considered as one of the most important wake problems from the engineering point of view. Little emphasis has been given to the boundary layer approximation. For turbulent flow,

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only semi-empirical theory has been discussed, and then briefly, and little has been said about the fields of turbulent fluctuations. The interesting problem of free boundary flow of inhomogenous fluid which is important in combustion problem has not been mentioned.

The main contents of the chapters are: Background and prospectus, circular sector hodographs, simple flows past wedge, general theory, multiple plates, curved obstacles, existence and uniqueness, compressibility and gravity, effective computations, axially symmetric flows, unsteady potential flows, steady viscous wakes and jets, periodic wakes, turbulent wakes and jets, miscellaneous experimental facts.

The book may be considered as an excellent review of potential flow problems with free boundary. In addition, some brief discussions of other free boundary flow problems from the mathematical point of view are given and these should be of value to students of fluid dynamics.

Guided Missiles in War and Peace, by Nels A. Parson Jr., Harvard University Press, Cambridge, Mass., 1956, 165 pp. $3.50.

Reviewed by VINCENT J. CUSHING
Illinois Institute of Technology

The book reviewed here is for the layman, and it is doubtful whether it would present new concepts or information to the guided missile engineer and scientist. However, it meets very well its avowed purpose of educating and orienting the layman and perhaps of serving as a source book of terminology.

Maj. Parson begins by undertaking a description of the principle of action and reaction as well as pertinent facets of aerodynamics. He then discusses thermal reaction engines from pulsejet through bipropellant rockets.

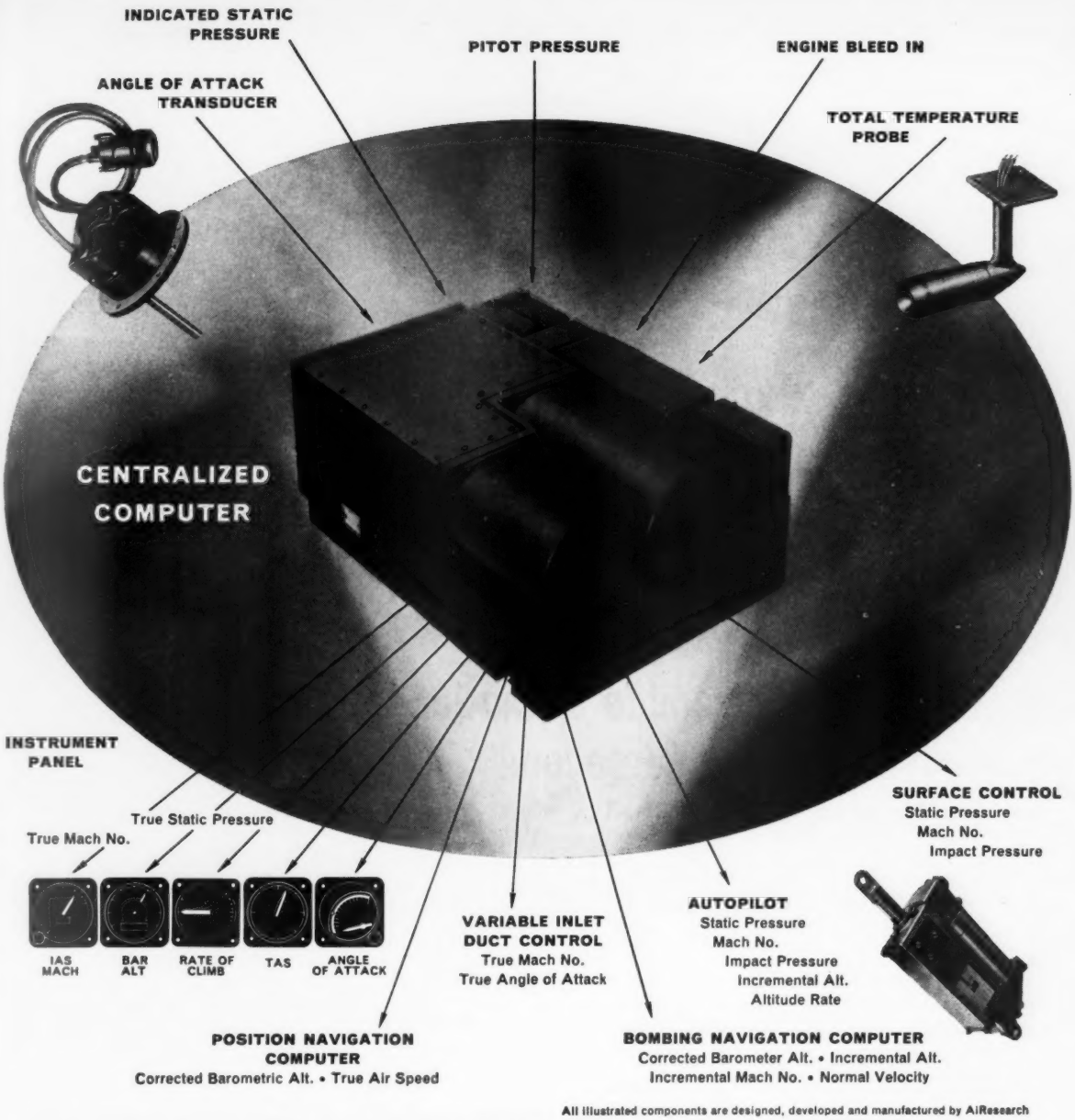
The author has also provided a suitable and interesting history of reaction engines extending from the Sinic first appearance of rockets through the current thinking in regard to proposals for interplanetary vehicles.

Schemes of guidance and control are discussed. Classical ballistic weapons are described and it is indicated how the advent of modern tactical conditions has pointed out basic limitations in such weapons. It is made evident that there exists a need for some sort of control of a missile after it has been launched, and discussions and descriptions are made of various classifications of guidance systems and their elements (e.g., passive seeker, command guidance, etc.).

Military applications are indicated and discussed for almost all categories of guided weapons: Surface-surface, surface-air, air-surface, air-air, underwater, etc

The author concludes with a discussion of hopeful commercial and peacetime uses for guided missiles, and applications of the attendantly evolved technical disciplines that have been prompted by the past and current need for guided missiles.

The reviewer feels that in a qualitative book such as this there might well have been included a discussion of current problems and expected obstacles in future developments. Obvious considerations would be cost, complexity and reliability.



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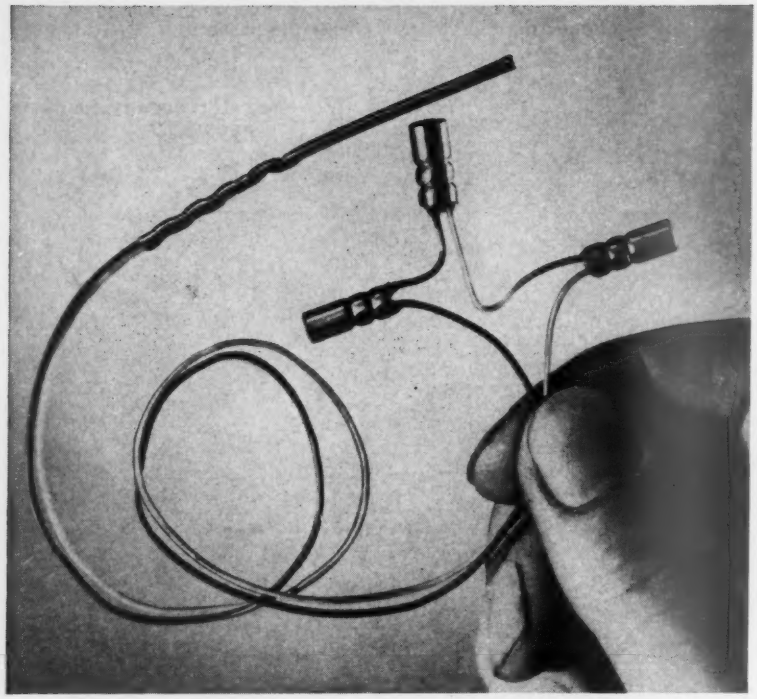

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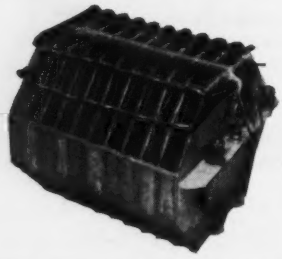

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Book Notices

Gas Dynamics, by Klaus Oswatitsch, translated by G. Kuerti, Academic Press, 1956, 610 pp. $12. This is a translation from the German of Klaus Oswatitsch's well-known text, "Gas Dynamics." Not only does this version represent the fine qualities of the original book but the various refinements reflect the competence of Gustav Kuerti who translated it. Excellently suited as a reference book or for instruction on the graduate level.

Modern Mathematics for the Engineer, Edwin F. Beckenbach, Editor, McGraw-Hill, 1956, 514 pp. $7.50. This book is an outcome of invited lectures at the University of California (Los Angeles). The topics covered and their respective authors are:

Linear and Nonlinear Oscillations, by Solomon Lefschetz.
Equilibrium Analysis: The Stability Theory of Poincare and Liapunov, by Richard Bellman.
Exterior Ballistics, by John W. Green.
Elements of the Calculus of Variations, by Magnus R. Hestenes.
Hyperbolic Partial Differential Equations and Applications, by Richard Courant.
Boundary-value Problems in Elliptic Partial Differential Equations, by Menahem M. Schiffer.
The Elastostatic Boundary-Value Problems, by Ivan S. Sokolnikoff.
The Theory of Prediction, by Norbert Wiener.
The Theory of Games, by H. Frederic Bohnenblust.
Applied Mathematics in Operations Research, by Gilbert W. King.
The Theory of Dynamic Programming, by Richard Bellman.
Monte Carlo Methods, by George W. Brown.
Matrices in Engineering, by Louis A. Pipers.
Functional Transformations for Engineering Design, by John L. Barnes.
Conformal Mapping Methods, by Edwin F. Beckenbach.
Nonlinear Methods, by Charles B. Morrey Jr.
What Are Relaxation Methods? by George E. Forsythe.
Methods of Steep Descent, by Charles B. Tompkins.
High-speed Computing Devices and Their Applications, by Derrick H. Lehmer.

Engineering as a Career, by Ralph J. Smith, McGraw-Hill, New York, 1956, 365 pp. $4.75. The primary objective of this book is to acquaint beginning students with the opportunities and responsibilities of an engineering career. It outlines the qualifications and duties of engineers and describes the type of work which is involved in the various branches of engineering. Previews of the fundamentals of mechanics, thermodynamics, electrodynamics and engineering economics are included. High school and college student advisers will find this book very useful for guidance work among pre-engineering students.

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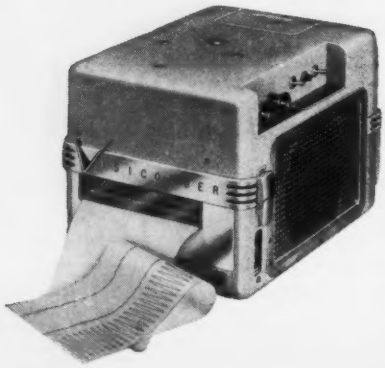

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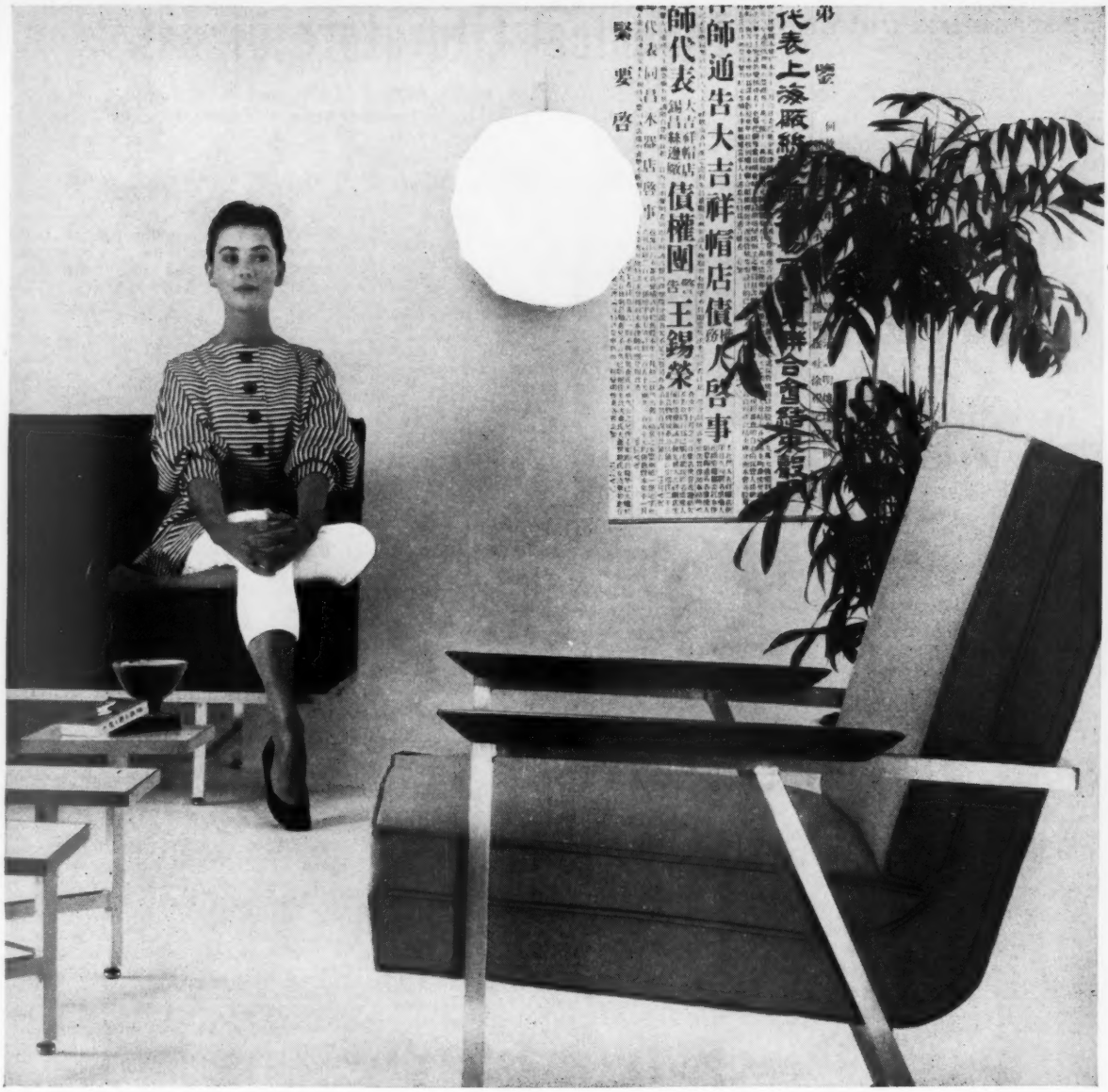

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Jet and Rocket Propulsion Engines

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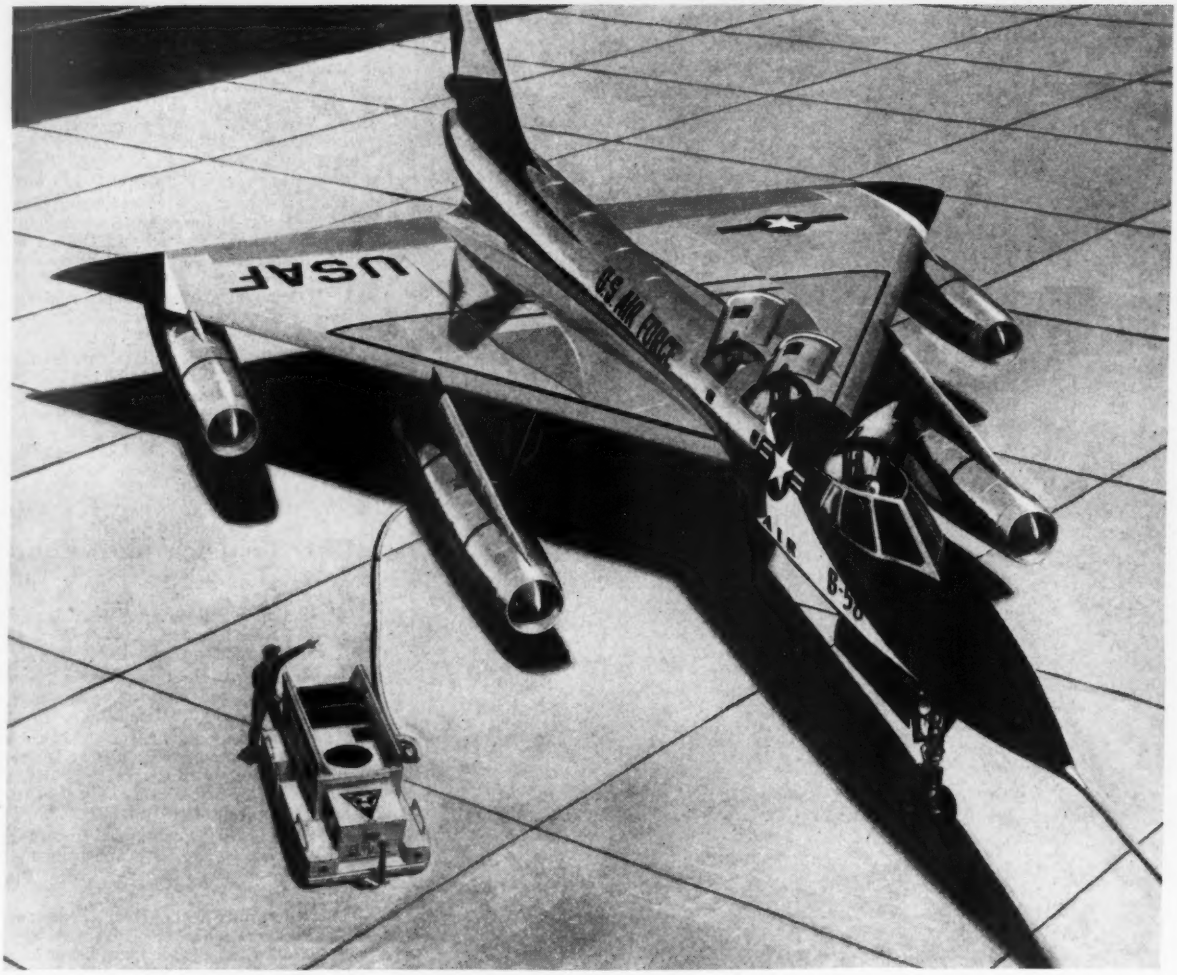
EDITOR'S NOTE: Contributions from E. R. G Eckert, J. P. Hartnett, T. F. Irvine Jr. and P. J. Schneider of the Heat Transfer Laboratory, University of Minnesota, are gratefully acknowledged.



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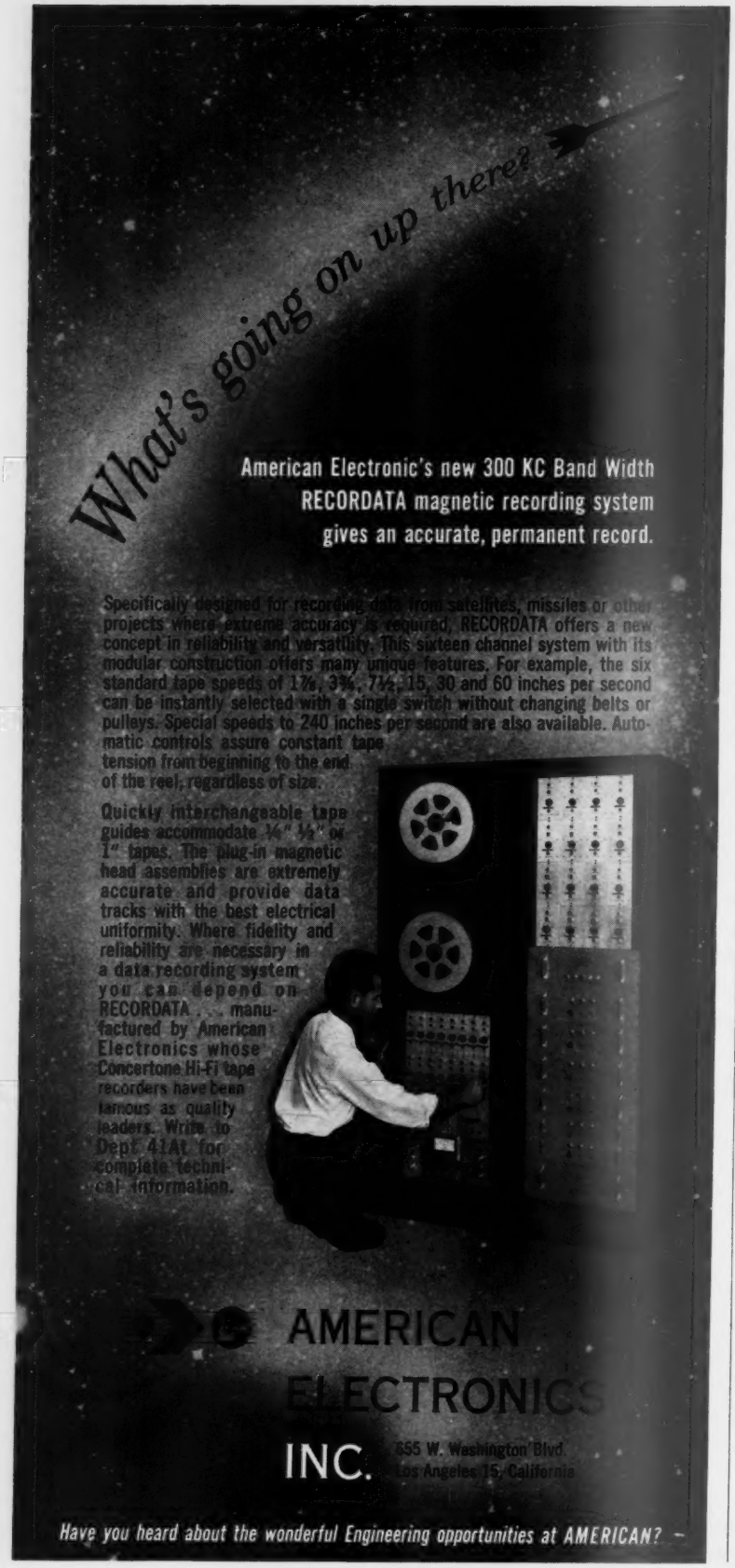
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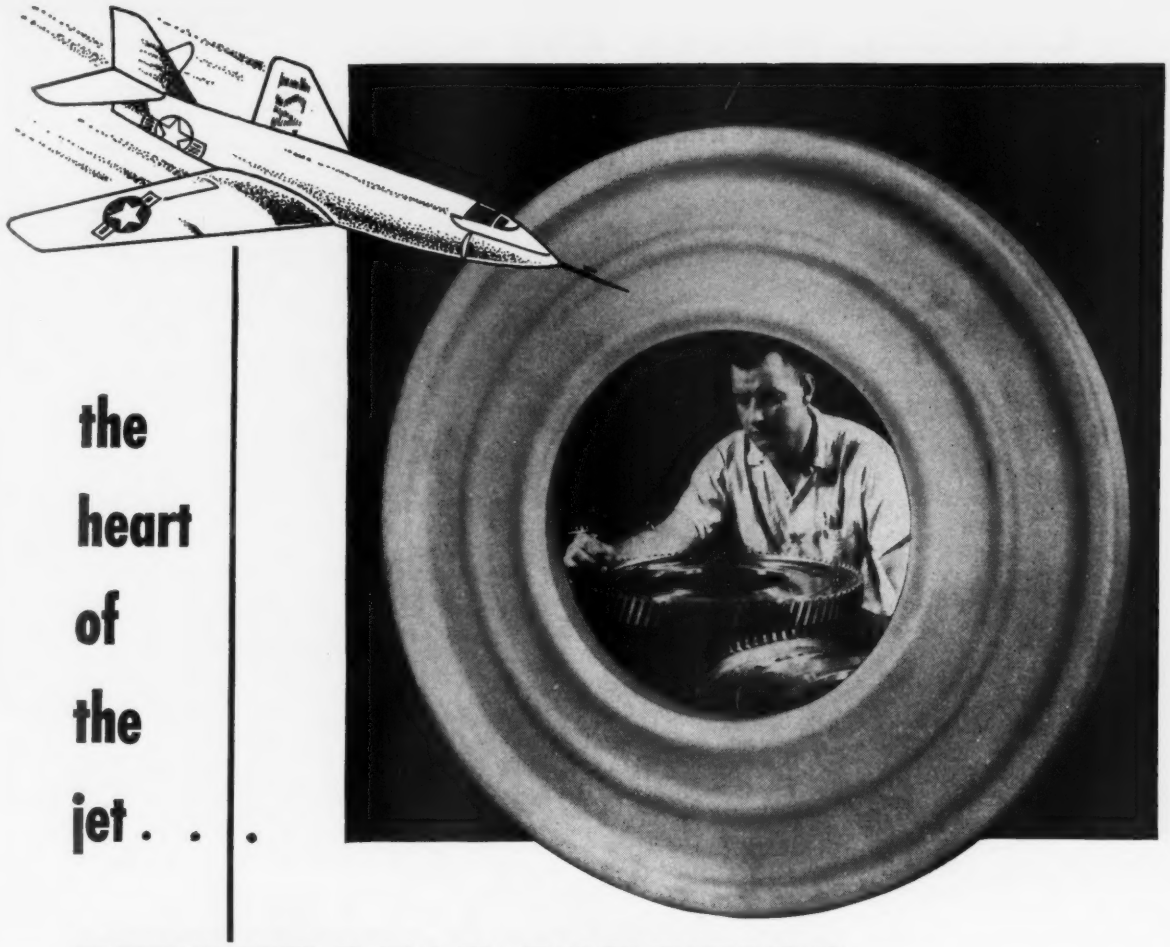

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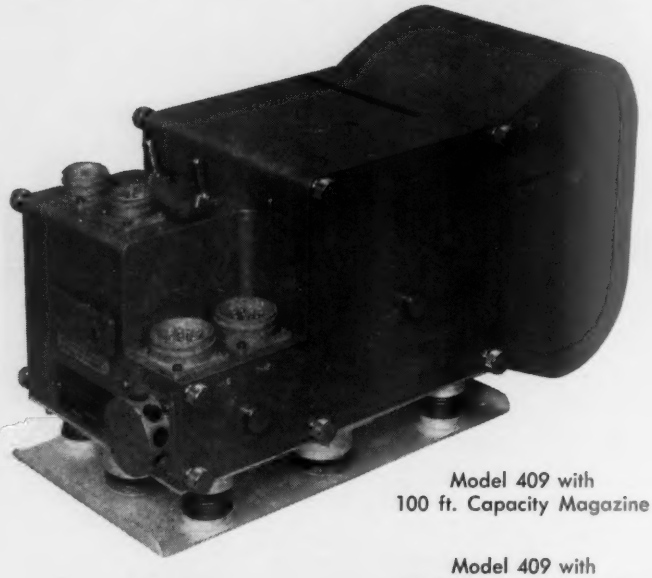

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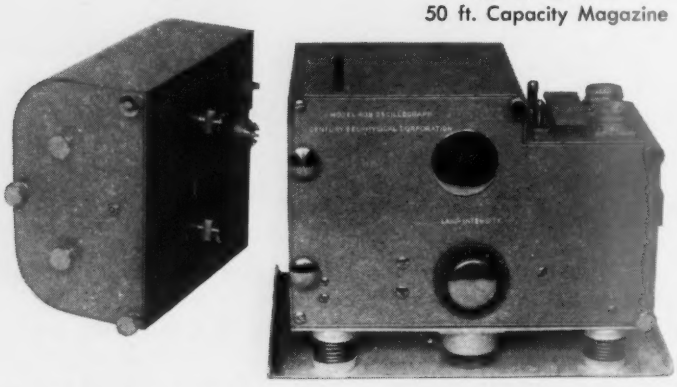

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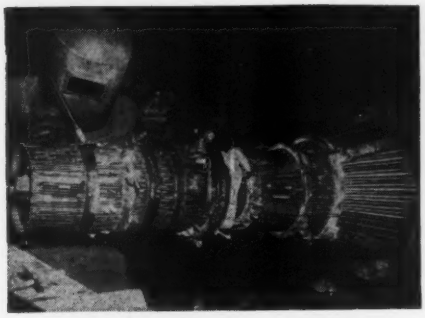

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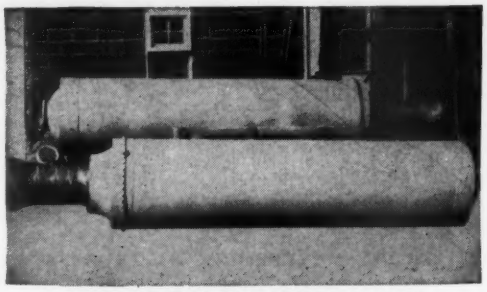

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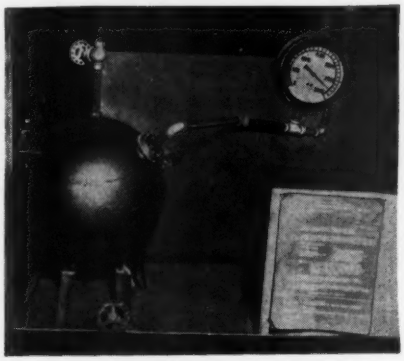

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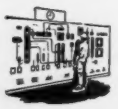
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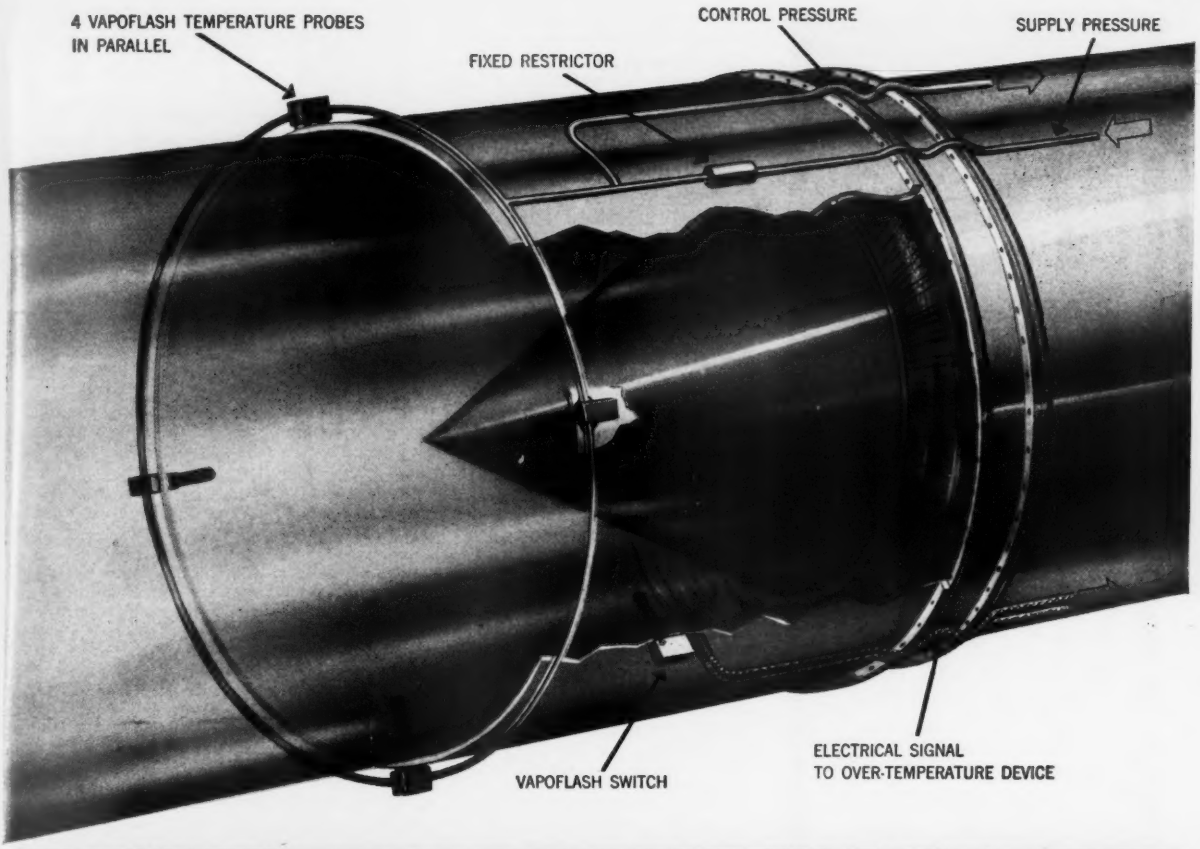

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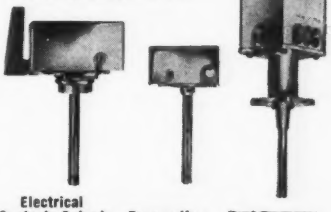

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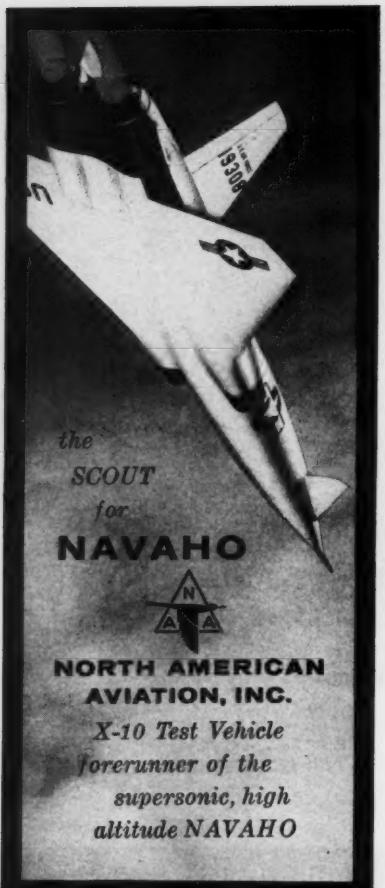

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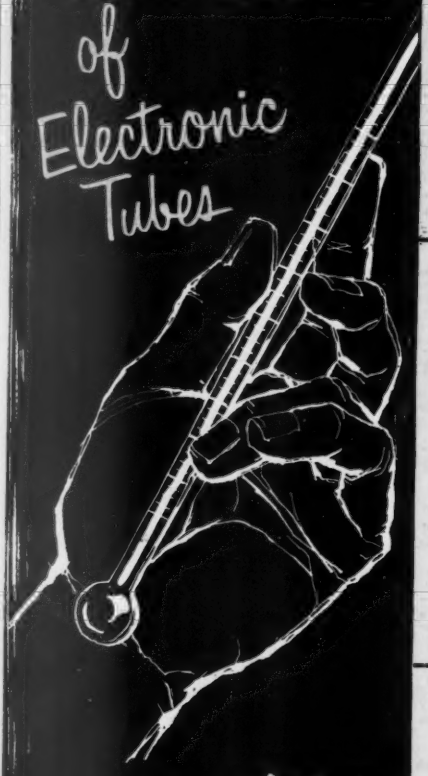

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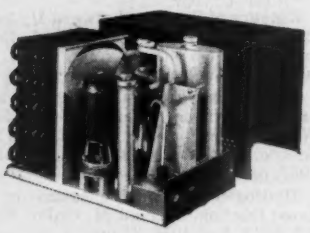

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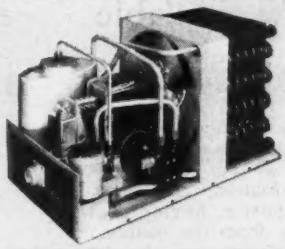

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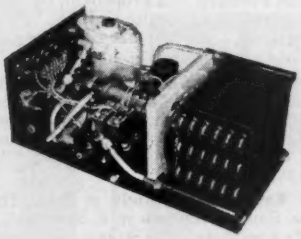

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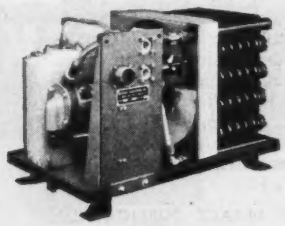

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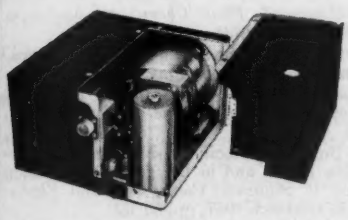

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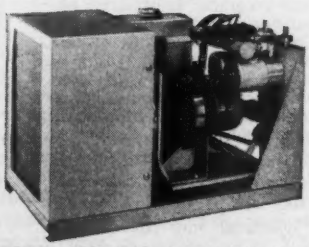

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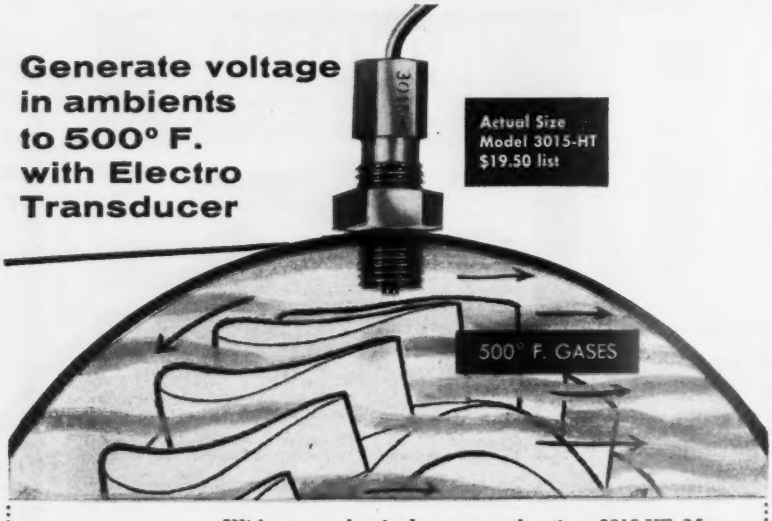

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Fig. 1

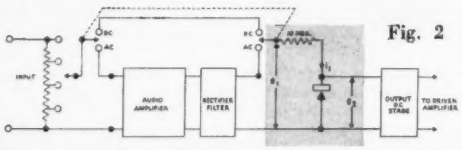


Fig. 2

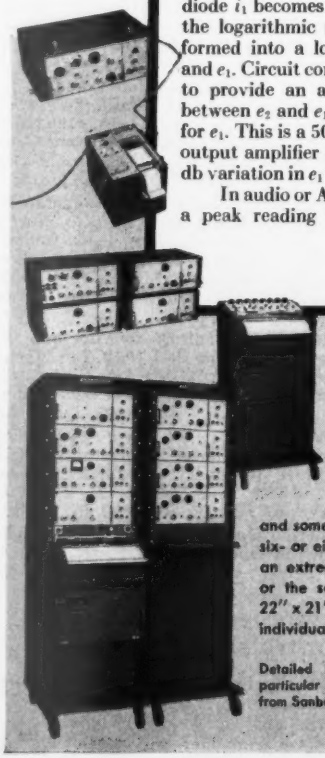
In audio or AC measurements, e_1 is derived from a peak reading type rectifier-filter circuit, which follows a high quality 20 cycle — 20 KC audio amplifier. With an input of 100 mv RMS, this amplifier will produce a 200 volt output from the rectifier. The 50 db chart, therefore, corresponds to a variation in AC input voltage of 0.316 to 100 mv.

In DC measurements, the audio amplifier is bypassed and the input applied to the diode circuit. Since the diode itself is a rectifier, used in the forward direction with its cathode near ground, the DC input must be polarized with the high side positive.

One broad area of application for the Log Audio preamplifier is audio level recording. For example, room reverberation time can be measured by recording sound level decay after the sound source is suddenly turned off, the reverberation time considered the period required for a 60 db decay to occur. Another example of audio signal recording is the plotting of frequency response curves of audio equipment such as microphones, filters, loudspeakers, etc. A multi-channel recording system with appropriate filters also makes possible audio spectrum analysis.

A second major type of application of this Preamp is the recording of DC voltages on a db basis. If the signals are small, a chopper can be used to convert DC to AC, thus taking advantage of the Preamplifier's audio amplifier. With an impedance matching transformer added to such an arrangement, the system becomes a logarithmic DC millivoltmeter or logarithmic DC microammeter of extreme sensitivity. Such a device could be used for plotting the volt-ampere characteristic of a germanium diode, which might be very helpful in selecting matched pairs of diodes. Another possibility is plotting the output of a fixed gain radio receiver and linear detector to a db scale, to rapidly record antenna performance data.

A comprehensive discussion of the design and these applications of the Log Audio Preamplifier is contained in an article by Dr. Arthur Miller, Chief Electrical Engineer of Sanborn Company, published in the Sanborn RIGHT ANGLE. Copies are available on request.



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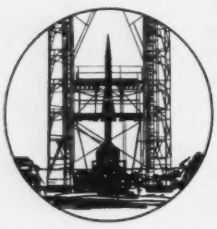

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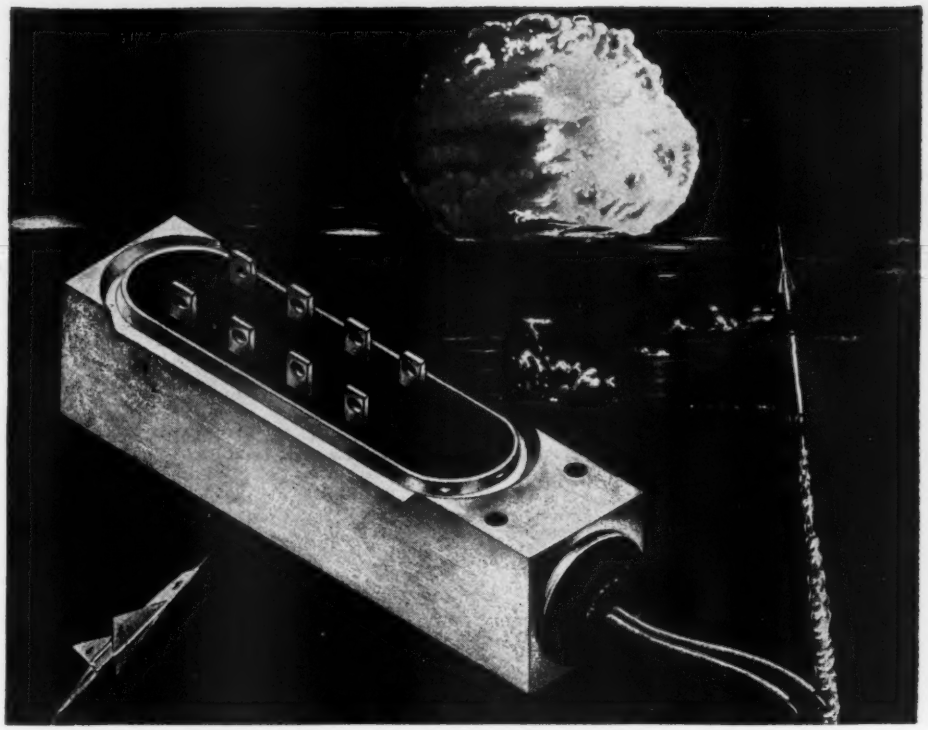

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Steady Nuclear Combustion in Rockets (Stationäre Kerverbrennung in Raketen), by E. Sänger, *NACA TM* 1405, April 1957, 39 pp. (Translation from *Astro. Acta*, vol. 1, pt. 2, 1955, pp. 61–88.)

Designing Electronics to Resist Nuclear Energy, by Harvey L. Morgan, *Electronics* vol. 30, May 1957, pp. 155–159.

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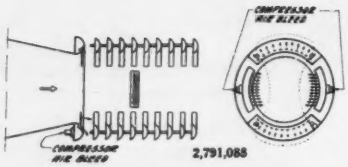


Device for thrust spoiling and thrust reversal (2,791,088). Gadicheria V. R. Rao, Cincinnati, Ohio, assignor to General Electric Co.

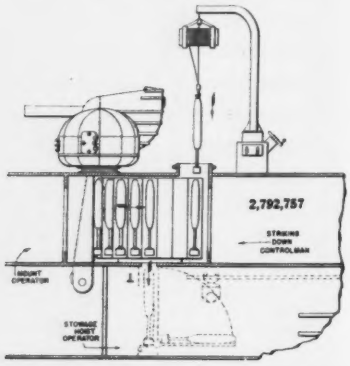
Boundary layer control consisting of an annular chamber surrounding the exhaust nozzle and openings for injecting air into the first stage of guide vanes to induce part of the exhaust gases to flow through the vanes. Lateral air injection cooperating with the boundary layer control deflects the gases into the vanes to produce reverse thrust.

Rocket igniter (2,791,962). Paul J. Terce, Paris, France, assignor to Society S.E.R.A.M.

Closure box containing the igniting charge, including a cup-shaped member with laterally extending U-shaped flange means closing the member and abutting against the wall of the rocket motor. The bottom wall of the member has perforations equidistantly spaced from the bottom wall and directed toward the charge.

Rocket launcher [design] (180,087). Robert F. Keuper, San Pedro, Calif.

Ornamental design of six-sided container with a sharklike nose, for carrying under the wing of an aircraft.



Missile launcher (2,792,757). Robert E. Carlberg, Frank W. Sieve and Rocco De Masi, Arlington, Va.

Automatic missile loading and launcher having a trainable mount with a guide for directing the initial flight of the missile. A hoist moves each missile in a vertical straight-line path into the guide.

Multi-cellular rocket package (2,792,962). Ernest H. Granfelt, Huntington Valley, Pa., assignor to the U. S. Navy.

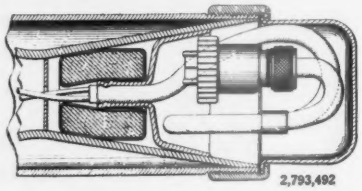
Lightweight high strength container and support for rockets. Polygonal hollow members open at both ends, consisting of thin sheet material, are grouped in a cylindrical cluster.

Jet pipe and jet deflecting nozzle device for a jet-propelled aircraft (2,791,089). Norman A. Dimmock, Woking, England, assignor to Power Jets (Research and Development) Ltd.

Top and bottom walls and side plates constituting a jet pipe. A nozzle wall is connected to the rear of the jet-pipe wall by hinges and slides linking together the two nozzle walls and constraining them to make corresponding angular movements.

Improved cooling and lubricating arrangement for bearings of a gas turbine engine (2,791,090). Stanley G. Hooker, Falfield, England, assignor to The Bristol Aeroplane Co., Ltd.

Liquid lubricant is supplied to flood the bearing chambers. Air is supplied to the chambers at a temperature lower than that of the air discharged from the compressor, and at a pressure higher than the lubricant.



Rocket assembly (2,793,492). Bruce H. Sage and Kenneth H. Rolinson, Pasadena, Calif., assignors to the U. S. Navy.

Flexible element for sealing a rocket nozzle. The element has a reduced neck and an enlarged body wedged into the exhaust port. The neck has an opening for passing and positioning an ignition cable in the nozzle.

Pulse jet heating burner control system (2,791,271). Theodore John Kauffeld, Convent, N. J.

Electrical preheater for a burner at starting. A primer injects fuel into the burner controlled by a time delay relay responsive to current in a circuit connecting the preheater to the power source.

Variable area nozzle for jet propulsion engine (2,791,882). Robert G. Willard, Wilkensburg, Pa., assignor to Westinghouse Electric Corp.

Circumferentially arranged elongated plates slidably mounted side by side on longitudinal tracks. Plates are interleaved to conform to the contour of the converging part of the discharge nozzle. Shifting the plates partly into or out of the nozzle opening varies the flow area.

Propellant system (2,791,883). George E. Moore and Frederic M. Cooper, Albany, N. Y., assignors to General Electric Co.

Fuel chamber in a rocket having solid fuel containing a hollow passage and a hydrogen peroxide decomposer chamber. Means are provided for contacting the fuel with the decomposition products of the peroxide.

Rocket projector (2,791,941). Chas. C. Lauritsen, Pasadena, Calif., assignor to the U. S. Navy.

Supporting guide with slits to accommodate an electrical contact engageable for energizing the rocket. A blast deflector plate having an inclined rear end diverging upwardly from the guide deflects blasts from the rockets.

Rocket tube muzzle door (2,792,754). Howard M. Walker, Torrance, Calif., assignor to Northrop Aircraft, Inc.

Launching tube in an aircraft. Parts of the leading edge directly ahead of the tube are set at an acute angle with respect to the axis of the tube. As the rocket is fired, a cover opens to permit free forward course for the rocket, after which the cover closes.

High speed launching device (2,792,755). Reinhard N. Lahde, Camarillo, Calif., assignor to the U. S. Navy.

Missile launcher with a cylinder closed at both ends, and having a longitudinal slot aligned with a tow fitting secured to a piston. As the piston moves in the cylinder, a sealing means is successively forced out of and back into engagement with the inner surface of the cylinder.

Fin arrangement for projectile (2,793,591). J. R. Jasse, Paris, France, assignor to Societe Nouville des Etablissements Brandt.

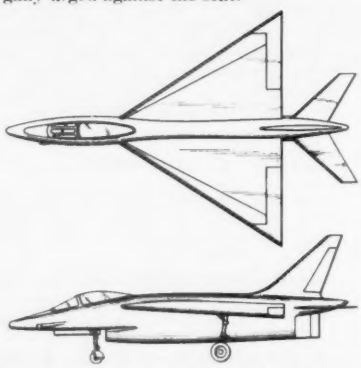
Pivotal fins having the general shape of the segment of a cylinder. They overlap one another in their inoperative position closed against a support fixed to the rear of the projectile.

Amphibious aircraft and elements thereof (2,793,827). Rudy F. Ries and Wm. R. Ryan, Riverside, Conn., assignors to Edo Corp.

Jet aircraft with a single main ski and steerable tail ski in tandem. Universally pivotable flexible wingtip skis hold the plane on an even keel when taxiing on ice, snow or other slippery surfaces.

Helicopter rotor ram jet drive (2,795,283). Edward L. Bennett, Palo Alto, Calif., assignor to Hiller Helicopters.

A by-pass valve in the line to the fuel supply. A spring urges closure away from the valve seat and inwardly relative to the rotor blade. Upon starting the engine and rotating the blade, the closure is centrifugally urged against the seat.



Aircraft design (180,297). Edgar Schmued and Welko E. Gasich, Los Angeles, Calif., assignors to Northrop Aircraft, Inc.

Delta-wing jet aircraft with engine intake under the fuselage, aft of the retractable nose wheel.

EDITORS NOTE: Patents listed above were selected from the Official Gazette of the U. S. Patent Office. Printed copies of Patents may be obtained from the Commissioner of Patents, Washington 25, D. C., at a cost of 25 cents each; design patents, 10 cents.

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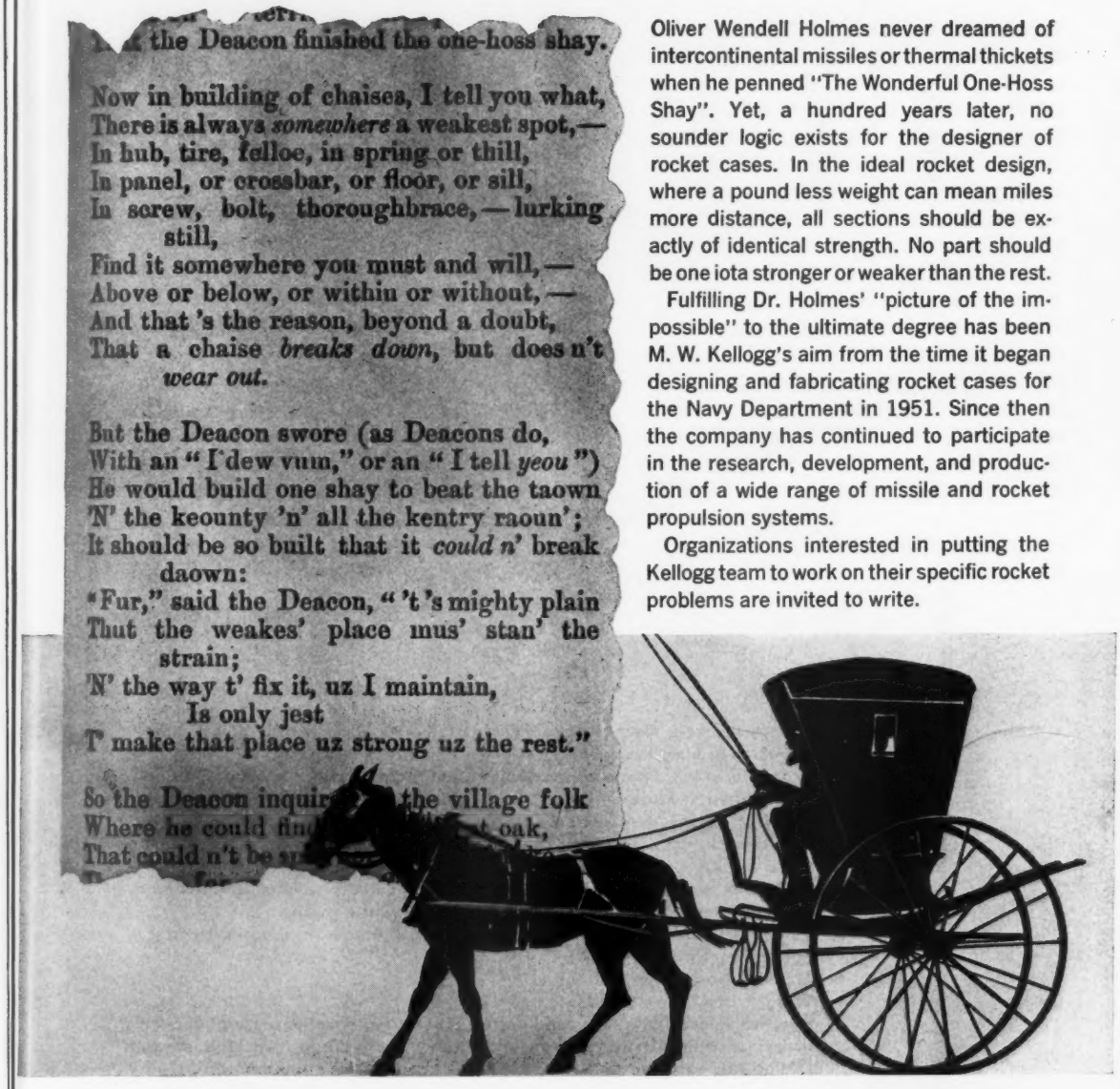
Rocket Logic in Retrospect



the Deacon finished the one-hoss shay.

Now in building of chaises, I tell you what,
There is always *somewhere* a weakest spot,—
In hub, tire, felloe, in spring or thill,
In panel, or crossbar, or floor, or sill,
In screw, bolt, thoroughbrace,—lurking still,
Find it somewhere you must and will,—
Above or below, or within or without,—
And that 's the reason, beyond a doubt,
That a chaise *breaks down*, but does n't *wear out*.

But the Deacon swore (as Deacons do,
With an "I dew vum," or an "I tell *yeou*")
He would build one shay to beat the taown
'N' the keounty 'n' all the kentry raoun';
It should be so built that it *could n'* break daown:
"Fur," said the Deacon, "'t 's mighty plain
Thut the weakes' place mus' stan' the strain;
'N' the way t' fix it, uz I maintain,
Is only jest
T' make that place uz strong uz the rest."

So the Deacon inquir[illegible] the village folk
Where he could find [illegible] oak,
That could n't be s[illegible]

Oliver Wendell Holmes never dreamed of intercontinental missiles or thermal thickets when he penned "The Wonderful One-Hoss Shay". Yet, a hundred years later, no sounder logic exists for the designer of rocket cases. In the ideal rocket design, where a pound less weight can mean miles more distance, all sections should be exactly of identical strength. No part should be one iota stronger or weaker than the rest.

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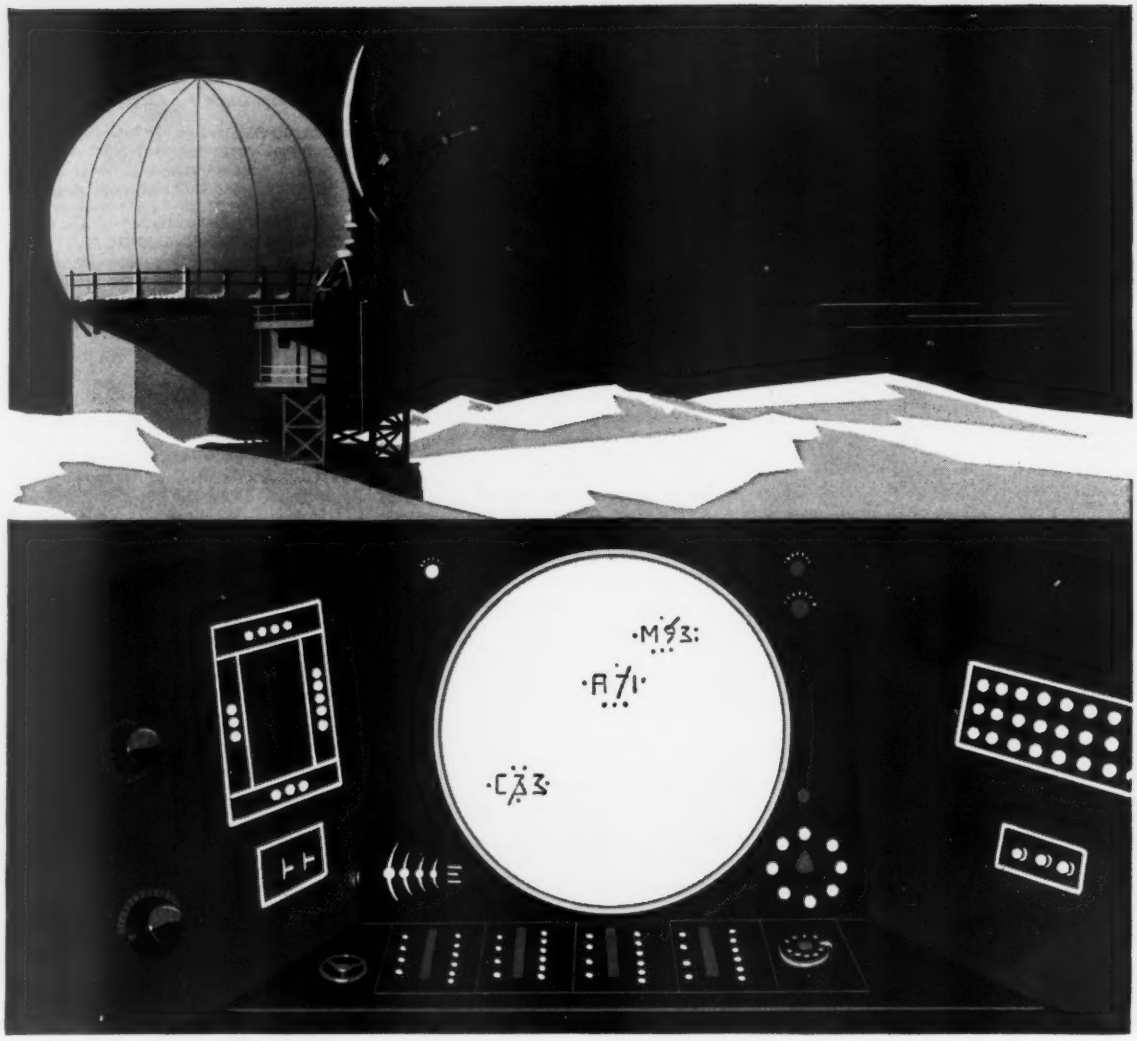

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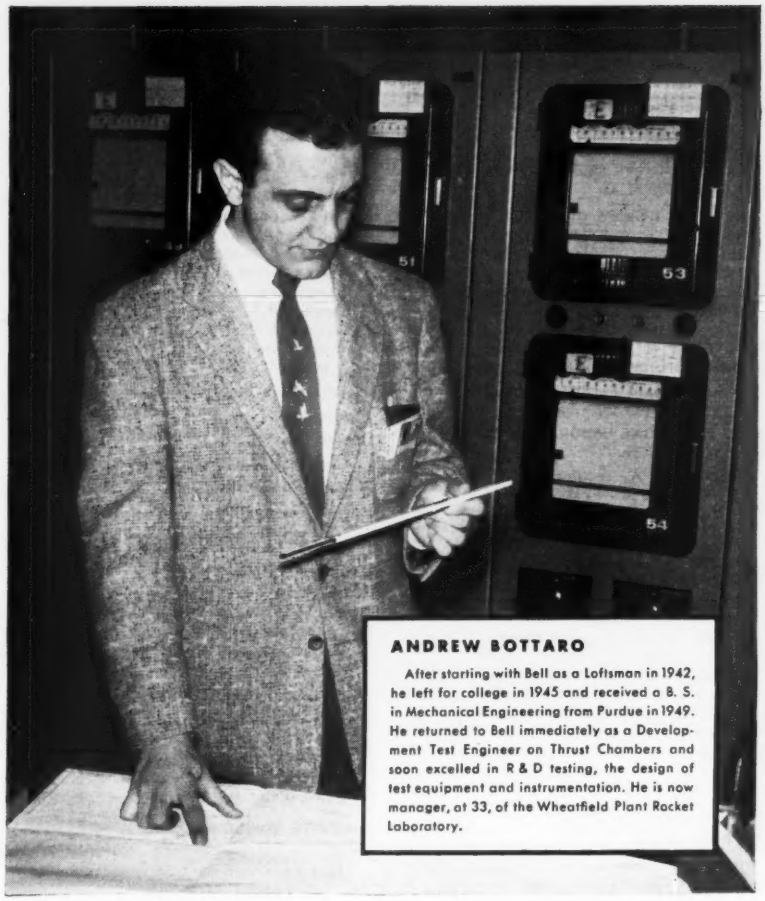

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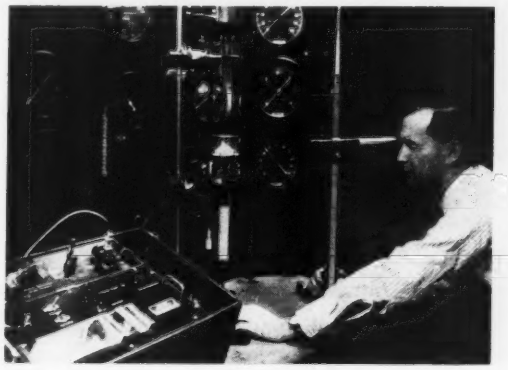

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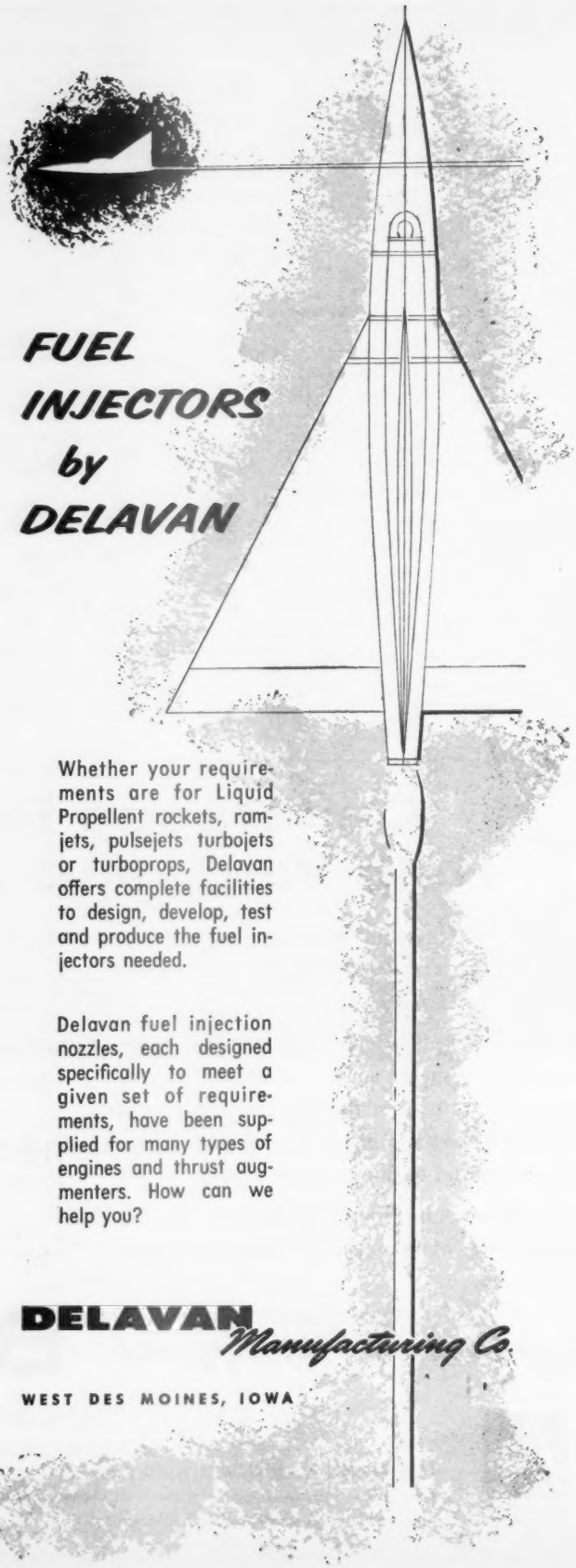
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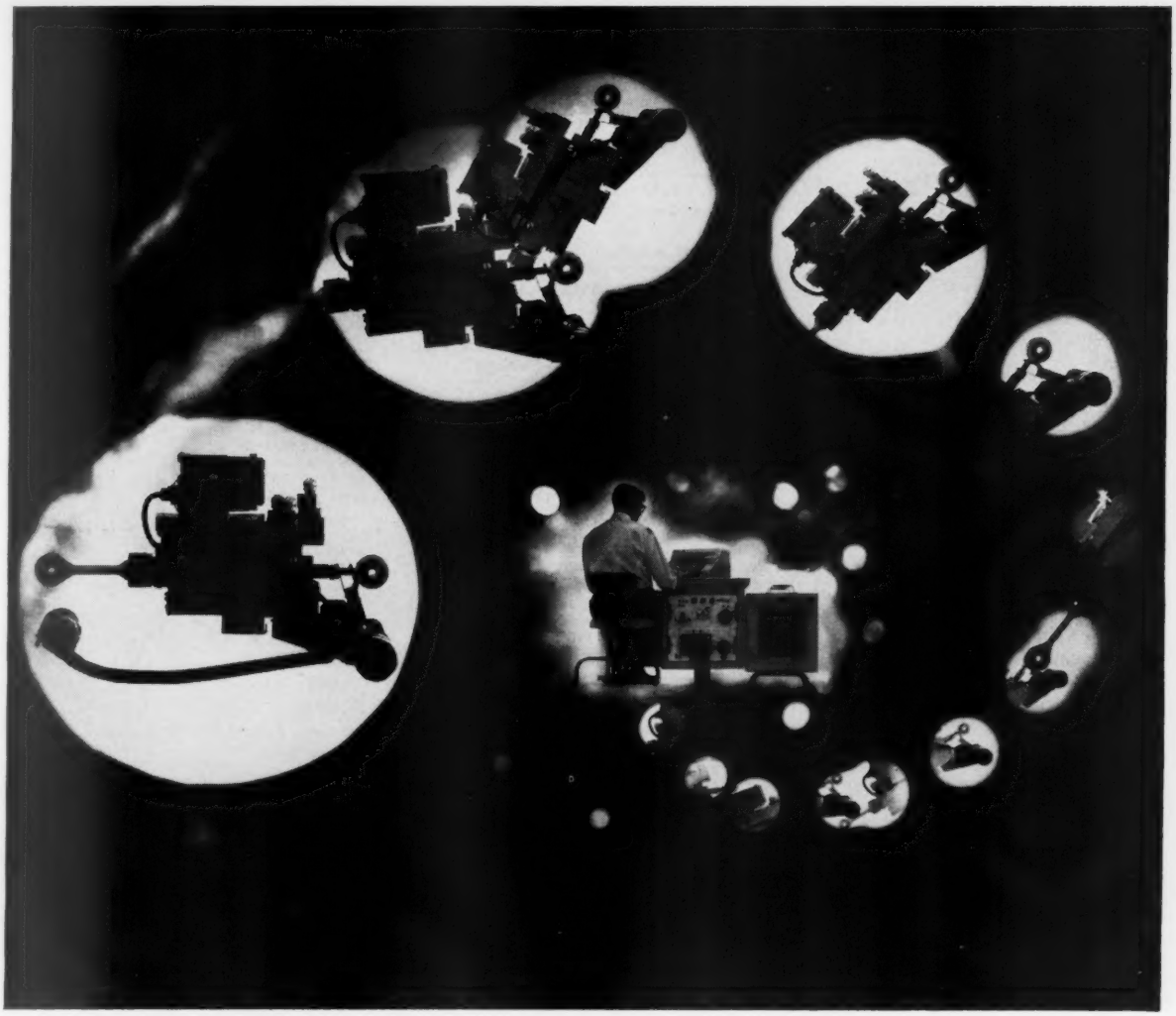

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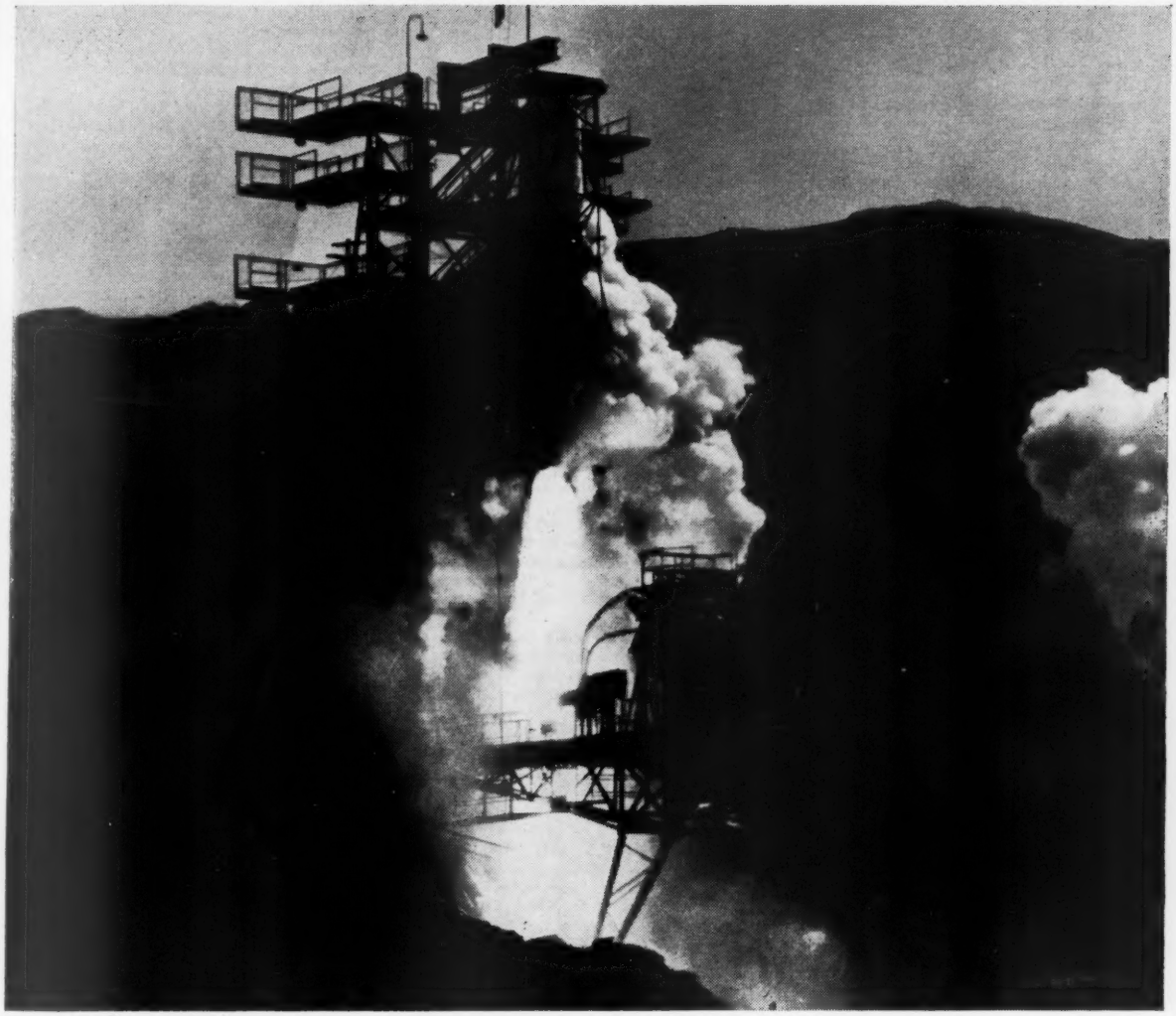

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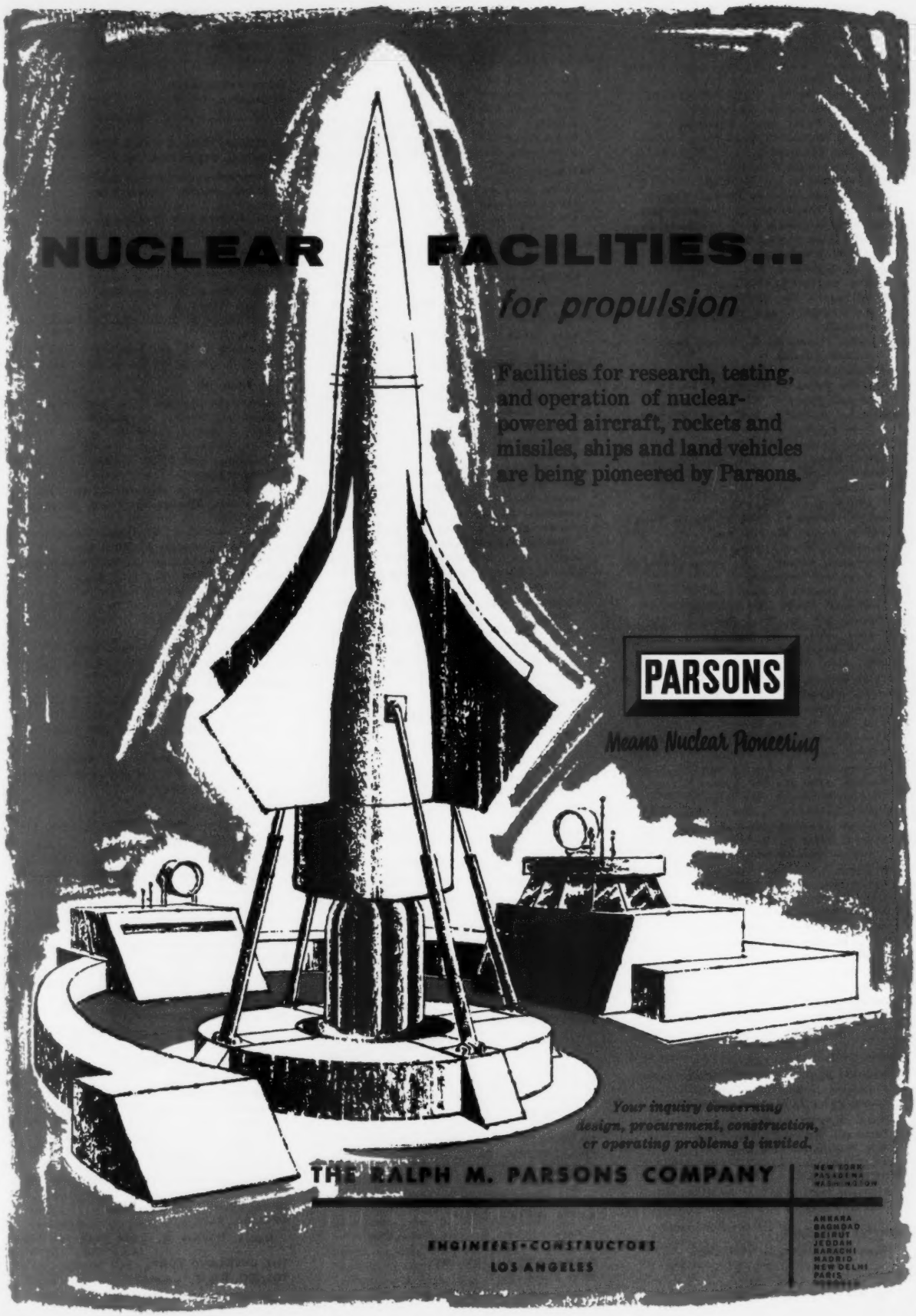
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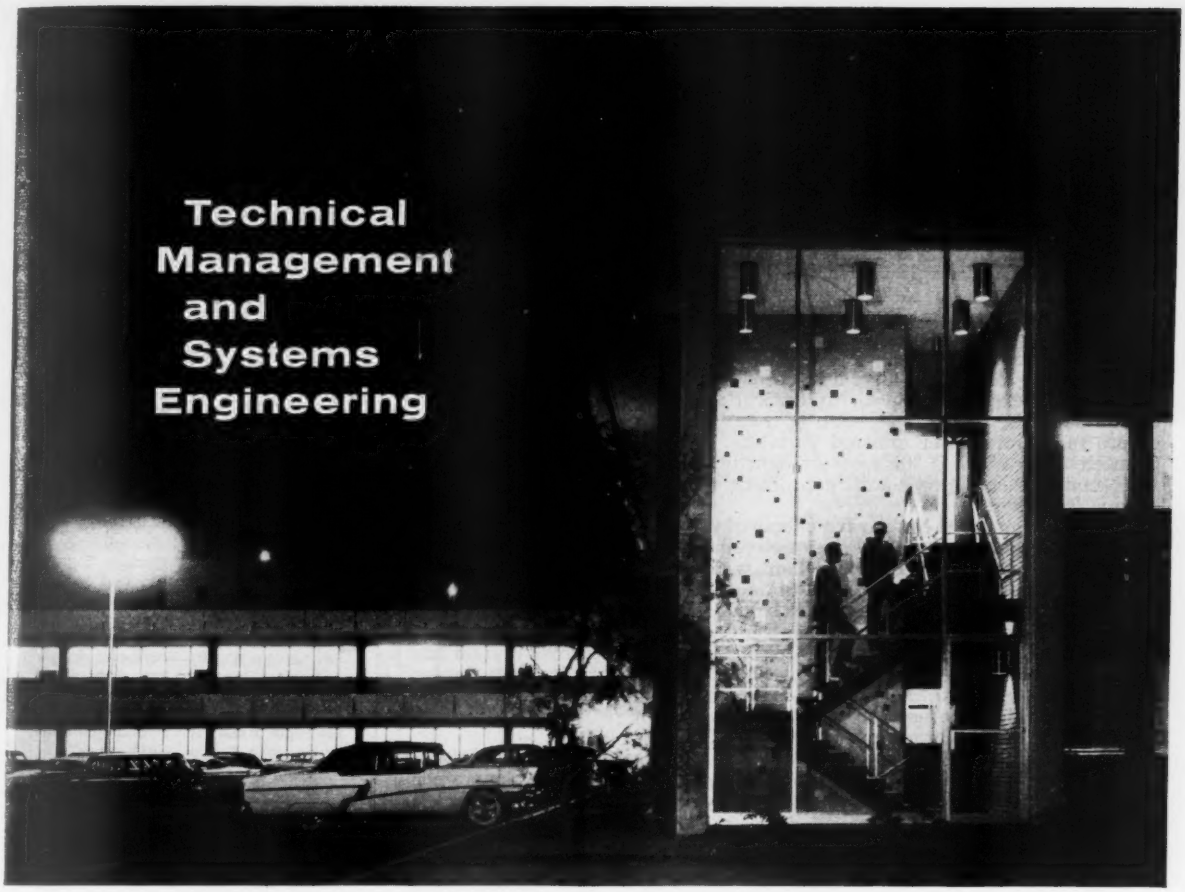

In systems engineering work, it is necessary to bring together a team that includes scientists and engineers of a wide range of technical specialties. In major weapons-systems projects, such teams will include hundreds of scientists and engineers.

But the assembly of a large group of scientists and engineers, no matter how capable they may be individually, does not of itself ensure good systems-engineering performance. The caliber of the project management has a major effect upon its technical accomplishment. It is not easy to coordinate the activities of large numbers of scientists and engineers so as not to stifle their creativeness on the one hand, nor to permit the various development sub-efforts to head toward mutually incompatible objectives on the other.

Of primary importance for good systems management is the philosophy underlying the selection of the supervisory personnel. The head of a technical activity should, first of all, be a competent scientist or engineer. A common mistake – nearly always fatal in systems work – is to fill such positions by non-technical men who have been trained only in management techniques. In the highly complex activities of major systems work, what is required is *technical management*, and of the two words, the word *technical* must never be overlooked.

In the selection of scientists and engineers for technical management, it is essential that the men chosen be broad in their training and approach. Each principal department head, for example, must have a good basic understanding of the technical facts of life of the other departments. When these people get together they need to speak a common language and understand each other's fields, so that proper decisions can be made on the many interrelated problems that come up. The higher the organizational responsibility of a technical manager, the more important this factor becomes.

The Ramo-Wooldridge Corporation is engaged almost entirely in systems work. Because of this, the company has assigned to scientists and engineers more dominant roles in the management and control of the business than is customary or necessary in most industrial organizations.

Scientists and engineers who are experienced in systems engineering work, or who have specialized in certain technical fields but have a broad interest in the interactions between their own specialties and other fields, are invited to explore openings at The Ramo-Wooldridge Corporation in:

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Atomic Radiation for Rocket Testing

by Alfred Cohen

Assistant Project Engineer in the Instrumentation Section of the RMI Test Department, Mr. Cohen is investigating new concepts for acquiring, reducing and evaluating test data. After graduating in mechanical engineering from City College of New York, he worked as a civilian test engineer for the Navy. Mr. Cohen came to Reaction Motors in 1954 as a Test Department Instrumentation Engineer.

Although the rocket engine in theory is an extremely simple powerplant, its development is a highly critical art. Successful operation of an engine depends upon the solution of many unique problems and the accurate measurement of unusual variables. One of the problems we encountered at RMI was the measurement of the corrosive effects of a new high energy rocket propellant, a problem we solved through the use of atomic radiation.

In determining the corrosive effects of propellants, it is essential that the sensing element employed does not come into direct contact with the propellant being tested. The propellant would rapidly damage or destroy most measuring equipment. In addition, the tolerance of the tests does not permit additional contamination from other foreign bodies.

We approached the problem on the basis that the density variation of the propellant could be the measure of its degree of corrosion upon the component materials with which it had been in contact . . . and the denser the material being tested, the less radiation would pass through it. We decided, therefore, to use the penetrating power of radiation as the tool to measure density variation.

To conduct the tests, a radioactive source and a detector for radiation were located so that the propellant being measured lay in a container between the two. The radioactive source emits gamma rays which penetrate through the propellant and its container. That part of the radiation that gets through the propellant and its container strikes the detector.

The detector contains a filling gas which is forcibly ionized when bombarded by atomic radiation. The energy of ionization is converted directly into electrical energy. Thus, an increase in the density of the propellant reduces the radiation field intensity at the detector, and results in a decreased generated signal.

However, accurate measurement of this energy is difficult because the generated signal is very low (in the magnitude of 1×10^{-13} amperes). To overcome this, an instrumentation system was specially developed to transmit, amplify and record this data.

The signal is transmitted to the amplifier by a special shielded cable that eliminates the effects of noise and other disturbances. Data is recorded by a Null system of measurement, using a ten-inch strip chart recorder. The recorder is calibrated to give full deflection for a density change equivalent to one tenth of a pound per cubic foot.

As a result of the data acquired by this test program, the research scientists have been able to improve the propellants to a point where they are both safe and highy desirable for use as rocket fuels.

This project is typical of one of the ways in which RMI Test Department supports the applied research, development and production projects associated with advanced rocket powerplants. All projects pass through this department, from the evaluation of components and fuels to the static "hot testing" of the entire engine system for an aircraft or missile before delivery.

The Test Department is staffed by a team of specialists highly qualified in mechanics, electronics, chemistry, and instrumentation. This team is constantly working to provide the most accurate possible test results and also to develop new equipment and methods that will allow for even more accurate information in the future. And—because this team is constantly expanding—we are always anxious to talk with top-rated engineers and technicians who are interested in test work.

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If you desire one or more reprints of Mr. Cohen's article, or would like to receive further information about employment at RMI, write to our Information Services Coordinator. Reaction Motors, Inc., 23 Ford Road, Denville, N. J.

Power for Progress

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ROCKET POWER PROGRESS REPORT

Taming Rocket Powerplants

by Walter O. Borcherdt

In charge of the Engineering Development Laboratory as well as the Valves and Controls Section at RMI, Mr. Borcherdt has over a decade of rocket experience. Before coming to RMI in 1951, he was senior project engineer on controls in the rocket department of Curtiss-Wright Corporation. Mr. Borcherdt received his degree in mechanical engineering from Stevens Institute of Technology in 1938.

Reliability of a rocket powerplant is the direct result of the simultaneously correct operation of all of its elements. Generally, this is assured by overcoming each element's own peculiar development problems. Among these elements is one type which has the interesting property of being composed of specific components, yet whose operational characteristics are significant only in the performance of the entire powerplant system. This element is the control system.

Present powerplant controls requirements are based on these straightforward, simple considerations:

1. **Prepare the powerplant for firing**
2. **Sequence and control the system to rated thrust**
3. **Vary thrust rapidly and safely as required**
4. **Shut down smoothly**

Step one may require the use of gas pressure regulators and relief valves to push the propellants safely from tankage to pumps. There must be switches and relays to indicate or control the valves which admit propellants to the gas generators and combustion chambers, and time delay cutouts to insure against unsafe accumulation of propellants.

In step two, ignition and combustion may be monitored to insure correct use of the high energy release rates. Turbines may be safely brought up to speed by throttling the propellants. Safety devices may be added to limit over-speed and avoid the danger of structural failure.

In step three, a controller (it may be hydraulic, electric, pneumatic, or a combination) is used to vary thrust. Here multiple loop servo systems are sometimes used to insure that thrust change rates, turbine speeds, vehicle acceleration or terminal velocity, and other parameters vary in a coordinated manner.

Finally, in multiple stage powerplants or piloted vehicles, the rocket engine must be shut down safely and the system automatically purged of propellant accumulations.

The successful implementation of these four steps involves the application of many devices of unique characteristics whose individual selection is intimately tied to the following question:

"What is the actual control system problem to be solved?"

Once this question is clearly and consistently defined, a tight performance envelope results. This makes the few possible solutions fairly obvious. The final stage is to apply the knowledge of a team of control specialists and proceed through the usual component development stages. At RMI, this team of specialists in controls engineering is made up of highly qualified mechanical, electrical and chemical engineers. These professional men have wide experience in the creation of control units and systems for rocket powerplants as well as other types of propulsion systems. However, there is a continuing need for qualified graduate engineers, able to participate in working closely with other groups of specialists, integrating the control system into the overall engine performance envelope.

The systems approach is of extreme importance to successful rocket powerplant control efforts. It is a standard approach of the RMI Controls Section. Sequence system design, switching circuit analysis, human engineering, feedback control methods, statistical procedures, information theory and latest research data on fluid flow control are among the techniques used. They are applied along with practical design concepts which result in the development of simple but effective controls systems.

If you desire one or more reprints of Mr. Borcherdt's article, or would like to receive further information about employment at RMI, write to our Information Services Coordinator. Reaction Motors, Inc., 23 Ford Road, Denville, New Jersey.

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Library & Info. Div., So. African Council for Scientific & Ind. Res., Pretoria
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SPAIN

Bustelo, Ramon, Madrid
C.E.T.M.E., Madrid
Distribuidora Internacional-Ceta, Madrid
Instituto Nacional de Tecnica Aeronautica, Madrid
Inst. Nacional de Tecnica Esteban Tradas Aeronautica, Madrid
La Sierra, Col. Manuel, Sevilla
Litavia, Madrid
Regimento de Artilleria de Costa Jefature de Instruccion, Algeciras
Sr. Ruy-Diaz-Litavia, Madrid
Talleres, B. K., Madrid

SWEDEN

A. B. Bofors, Central Library, Bofors
AB Volvo, Biblioteket, Goteborg
Backman, Torsten, Trollhattan
Biblioteket Marinforvaltningen, Stockholm
Chalmers Univ., The Library of Technology, Gothenburg
Flygtekniska Forsoksanstalten, Ulvsunda
Forsvarets Fabriksverb Akers, Akers
Forsvarets Fabriksverk, Carl Gustafs Stads Gevarsfaktori, Eskilstuna
Forsvarets Fabriksstyrelse, Stockholm
Forsvarets Forskningsanstalt, Stockholm
Forsvarets Forskningsanstalt, Sundbyberg
Gudmunsson, G., Trollhattan
Institutionen for Flygteknik, Stockholm
K. Tekniska Hogskolans Bibliotek, Stockholm
Kgl. Flygforvaltningen Robotvapenyran, Stockholm
Kingl Flyforvaltningen Forsokscentralen, Biblioteket, Malmslatt
Kungl. Armetygforvaltningen, Stockholm
Kungl. Flygforvaltningen Materialavdelningens Bibliotek, Stockholm
Kungl. Krigshogskolan, Biblioteket, Stockholm
Kungl. Tekniska Hogskolan, Institutionen for Uppvarmnings Ventilationsteknik, Stockholm
Lindstahls Lilla Boklada, Stockholm
Lindstahls Lilla Boklada, Stockholm
Satmark, Lennart, Goteborg
Scandinavian Airlines System, Library, Stockholm
Svenska Aeroplan AB, Linkoping
Svenska Flygmotor A. B., Trollhattan

SWITZERLAND

Buvhhandlung fur Technik und Wissenschaft, Zurich
Eidg. Pulverfabrik, Wimmis Be
Halder, F., Bremgarten
Hispano Suiza S. A., Geneva
Pinkus & Co., Zurich
Zahn, Willy, Zuerich

TURKEY

Robert College, Istanbul

URUGUAY

Facultad de Ingenieria Biblioteca, Montevideo

USSR

Biblioteka, Moscow
Biblioteka Kombinata, Moscow
Cnt. Biblioteke, Priborostroenija, Moscow
CNTB Ministerstva Avtopromishlennosti, Moscow
Dom Uchenykh, Glavnyj Pochtamt, Moscow
Filial Biblioteki, Moscow
Gos. Nauchn Biblioteka, Minist. Vyssh. Obraz, Moscow
Gos. Nauchn. Biblioteka, Minist. Kultury SSSR, Moscow
Gosudarstvennaja Biblioteka SSR im. Lenina, Moscow
Gostekhnika SSSR, Moscow
Inoizdat Glavnij Pochtamt, Moscow
Inst. Mekhaniki Glavnyj Pochtamt, Moscow
Ministerstvo Aviapromyshlennosti, Moscow
Neftjanaja Biblioteka, Moscow
Technich Biblioteka Glavnyj Pochtamt Pochtov, Moscow
Tech. Biblioteka Glav. Upravlenia, Moscow
Vseojuznaja Gosbiblioteka Inoliteraturi, Moscow

YUGOSLAVIA

Jugoslovenska Knjiga, Beograd
Izdavacko Poduzece, Zagreb
Technicki Faultet, Zagreb
Trgovinska Preduzece Jugoslovenska Knjiga, Beograd

Please give the attached application form to someone you feel would be interested in ARS membership.

American Rocket Society, Inc.

500 FIFTH AVENUE • NEW YORK 36, N. Y.

APPLICATION FOR MEMBERSHIP

FOR OFFICE USE ONLY

No.

Adm.

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Cert.

To the Membership Committee:

I hereby apply for membership in the American Rocket Society, Inc. and enclose $_____ as my annual membership dues, $5.00 of which is for a year's subscription to the Journal of the American Rocket Society. The information given herewith is correct to the best of my knowledge.

Check type of membership desired (see other side for qualifications)

☐ MEMBER—annual dues $15.00

☐ ASSOCIATE MEMBER—annual dues $15.00

☐ STUDENT MEMBER—annual dues $5.00

Date _____

Signature in Full _____

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PLEASE PRINT OR TYPE

Name _____ Title (or military rank) _____

Organization or School _____

Services or Products of Organization _____

Your Duties and Responsibilities _____

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Organization Address ☐ } Check Address for Society Mail

Home Address ☐ }

Age _____ Date of Birth _____ Country of Birth _____ Citizen of _____

Other Technical Organization Memberships _____

_____

EDUCATIONAL RECORD

Secondary School _____ Date _____

College or University _____ Date _____

Major Subject _____ Degree Received _____ Date _____

Names of other Technical Schools or Colleges _____

Additional Degrees _____ By Whom Conferred _____ Date _____

REFERENCES

Name three and include addresses. Include at least one ARS member if possible.

_____

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(Over)

PROFESSIONAL RECORD Optional for Student Member Applicants. If space is inadequate add supplementary sheet of this size.

Dates: from to	Names and addresses of past employers. Your position, duties and responsibilities.

MEMBERSHIPS

MEMBER: Members shall consist of engineers and scientists who are actively engaged in the development or application of rocket or jet propulsion, other persons who have been working on the development or application of rocket or jet propulsion for at least four years and who hold or have held responsible positions in these fields, and such persons as may be deemed eligible for this class of membership by the Board of Directors by virtue of their outstanding accomplishments in other fields and their unusual interest in the purposes of the Society.

ASSOCIATE MEMBER: Associate members shall consist of persons, other than students, who are actively interested in the development or application of rocket or jet propulsion.

STUDENT MEMBER: Student members shall consist of persons not less than 17 years of age whose principal occupation is study at a recognized educational institution or who are serving as enlisted personnel in the Armed Forces of the United States, and who are interested in the development or application of rocket or jet propulsion.

CORPORATE MEMBER: Corporate Members shall consist of educational, scientific or industrial organizations who may choose this method of expressing their interest in the development or application of rocket or jet propulsion, and who are considered acceptable by the Board of Directors. Each Corporate Member shall be entitled to five representatives with the rights and privileges of Members. (Write for special Corporate Membership application form.)